Lunar Landing Research Vehicle

SUMMARY of ESTIMATED PERFORMANCE

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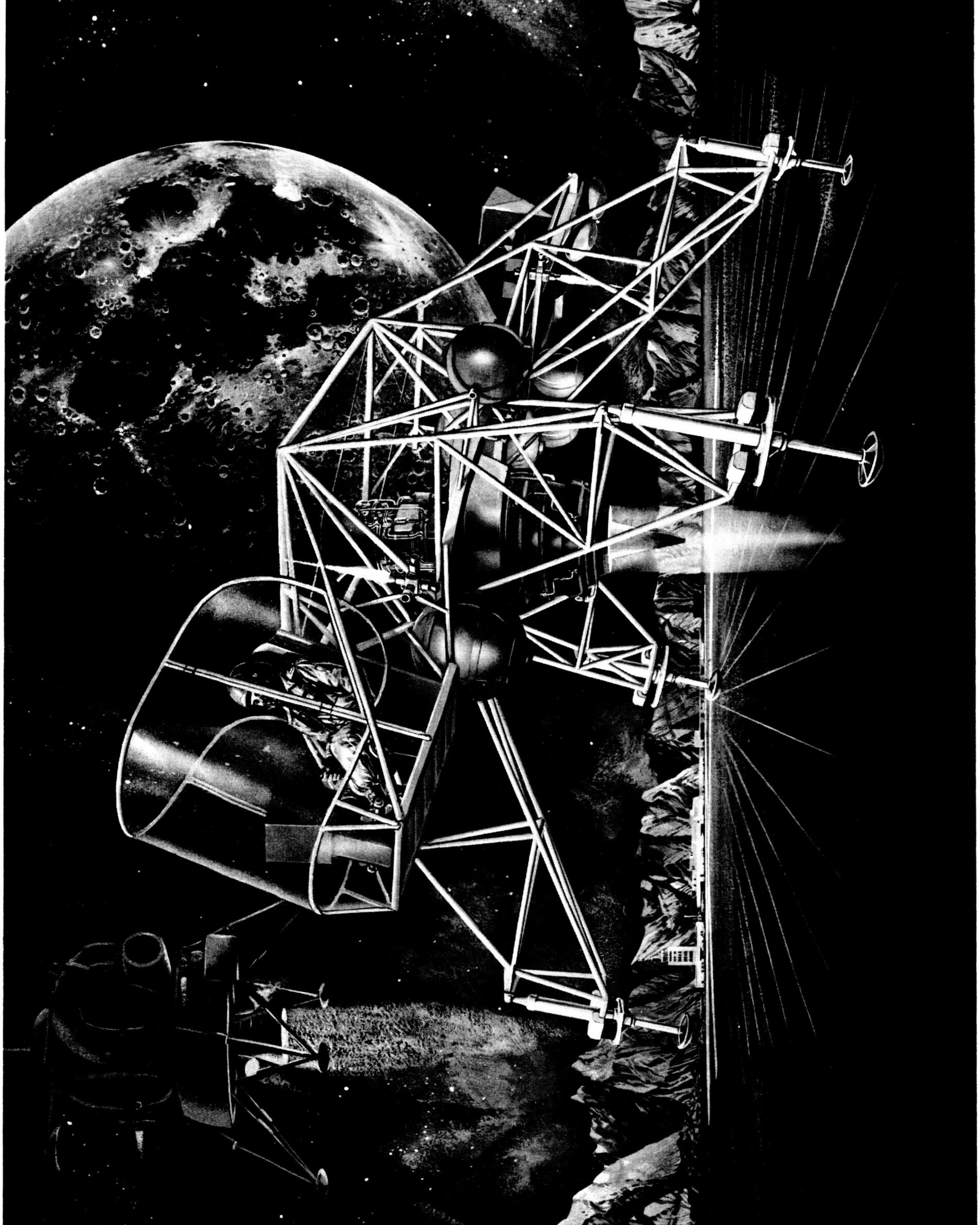
TABLE OF CONTENTS

LIST OF ILLUSTRATIONS

LIST OF TABLES

SUMMARY

This report summarizes the final results of the aerodynamic and performance analyses performed on the Lunar Landing Research Vehicle. Equations and/or graphs representing this data are presented along with brief descriptions of their use.



Lunar Landing Research Vehicle

SECTION I

INTRODUCTION

The Lunar Landing Research Vehicle (Figure 1-1) is a free flight simulator designed to explore the regions of lunar approach, hover, and touchdown. It uses a CF700-2V turbofan jet engine and throttleable hydrogen peroxide rockets for lift and propulsion. The lunar environment is simulated by removing the aerodynamic forces and moments and 5/6 of the earth weight of the vehicle from the pilot handling "feel." This is done using automatic compensation techniques. Primary force compensation is provided by the turbofan engine; moment compensation is through automatic control of the attitude rockets. These same attitude rockets are used by the pilot to control the vehicle attitude. During a lunar simulation the H_2O_2 lift rockets are used by the pilot to control the descent and, by tilting the vehicle, to control horizontal motions.

Aerodynamic forces and moments have been estimated and are presented in forms suitable for conversion to an analog computer setup. Performance parameters, primarily fuel consumed, have also been estimated. These are presented in a manner which facilitates the computation of the fuel required for any given mission.

Since the vehicle will be operational at NASA-FRC, Edwards AFB, the computations and graphs are designed for ease of use there. Ground level is taken as 2300 feet; standard day temperature is 50.8 degrees F.

SECTION II

BASIC DATA

2.1. AERODYNAMIC DATA

This section presents the results obtained from the various theoretical and experimental analyses of the LLRV. Results are presented primarily in the form of equations. Both the angular and the velocity-component forms of the equations are shown where feasible. The transformations required to obtain the jet engine aerodynamic parameters in the body-axis system are given. The angles and axes systems used are shown in Figures 2-1 through 2-3.

(1) Figure 2-1 gives the relation between the velocity vector and the X_V, Y_V, Z_V, body-axes. The pilot is facing in the positive X_V direction as per standard aircraft convention.

(2) Figure 2-2 shows the jet engine axes (X_e, Y_e, Z_e) and the gimbal angles θ_{jv} and ϕ_{jv}. The X_e axis is in the X_V-Z_V plane. The Z_e axis is along the center line of the engine and, with the Y_e axis, is in a plane rotated through the angle θ_{jv} about the Y_V axis. The angle ϕ_{jv} is the rotation about the X_e axis. The thrust vector is in the negative Z_e direction.

(3) Figure 2-3 shows the positive directions of the aerodynamic forces and moments.

2.1.1. Outer Frame Parameters. - Equations for the aerodynamic coefficients of the vehicle outer frame are given in Table 2-1. These coefficients were determined using methods developed from the subsonic cross flow principle. These methods were revised after comparisons with the wind tunnel data from the original "man-on-top" configuration and should provide accurate data on the current configuration.

Aerodynamic coefficients are based on a reference area, S, of 38.48 square feet and a reference length, l, of 7.0 feet. S is the area of a 7.0-foot diameter circle, which corresponded to the size of the pilot's platform in the original "man-on-top" configuration. The area has no physical aerodynamic significance, as a wing, would have for example, and therefore S was not changed when analyzing the newer configuration.

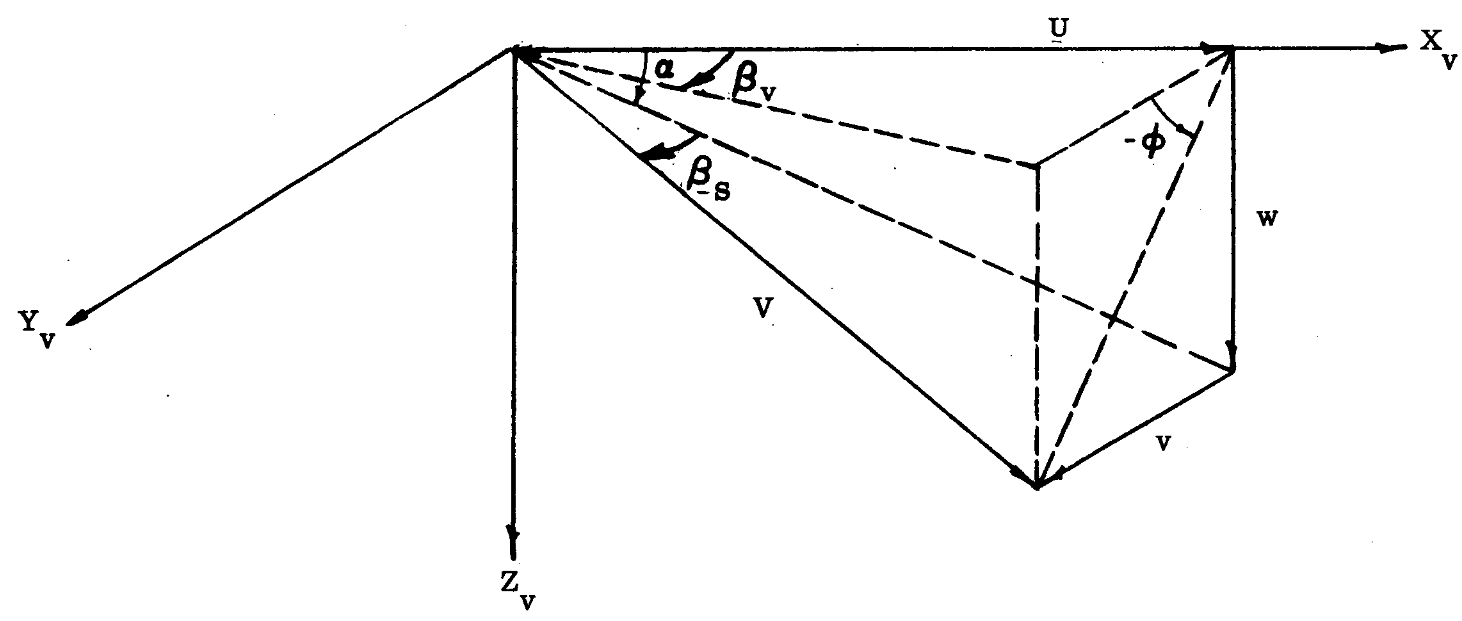
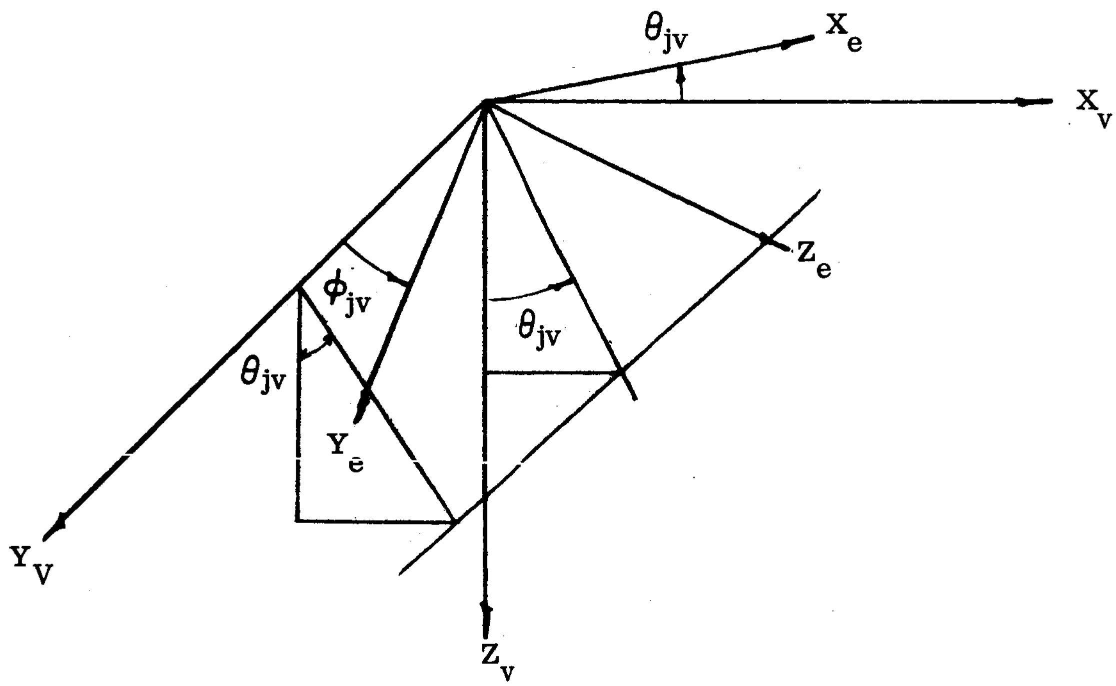


Figure 2-1. Definition of Aerodynamic Angles



$$\begin{pmatrix} \cos\theta_{jv} & 0 & -\sin\theta_{jv} \\ \sin\phi_{jv}\sin\theta_{jv} & \cos\phi_{jv} & \sin\phi_{jv}\cos\theta_{jv} \\ \cos\phi_{jv}\sin\theta_{jv} & -\sin\phi_{jv} & \cos\phi_{jv}\cos\theta_{jv} \end{pmatrix} \begin{pmatrix} X_v \\ Y_v \\ Z_v \end{pmatrix} = \begin{pmatrix} X_e \\ Y_e \\ Z_e \end{pmatrix}$$

Figure 2-2. Definition of Gimbal Angles

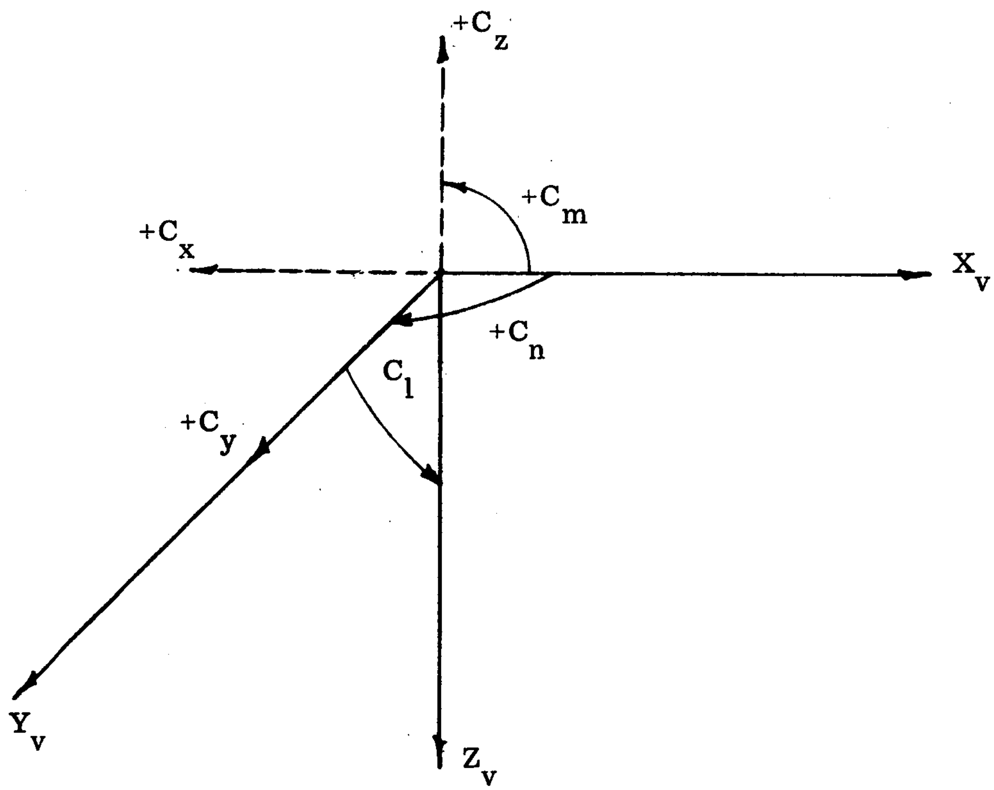


Figure 2-3. Aerodynamic Forces and Moments

TABLE 2-1

OUTER FRAME AERODYNAMIC PARAMETERS

S = Reference Area 38.48 ft^2
l = Reference Length 7.0 ft
$K_F = 1/2 \ \rho S$
$K_M = 1/2 \ \rho Sl$

COEFFICIENTS

$$C_x = 1.965 \cos \alpha = 1.965 \ u/\sqrt{u^2 + w^2}$$
$$C_y = 1.709 \sin \beta_v = -1.709 \ v/\sqrt{u^2 + v^2}$$
$$C_z = 1.830 \sin \alpha = 1.830 \ w/\sqrt{u^2 + w^2}$$

a. Equipment Platform Forward (150 lb Pilot)

$$C_m = -0.1052 \cos \alpha = -0.1052 \ u/\sqrt{u^2 + w^2}$$
$$C_n = -0.361 \sin \beta_v = -0.361 \ v/\sqrt{u^2 + v^2}$$
$$C_l = -0.158 \cos \phi = -0.158 \ v/\sqrt{v^2 + w^2}$$

b. Equipment Platform Aft (200 lb Pilot)

$$C_m = -0.129 \cos \alpha = -0.129 \ u/\sqrt{u^2 + w^2}$$
$$C_n = -0.340 \sin \beta_v = -0.340 \ v/\sqrt{u^2 + v^2}$$
$$C_l = -0.172 \cos \phi = -0.172 \ v/\sqrt{v^2 + w^2}$$

FORCES AND MOMENTS

$$F_{x_v} = K_F \ C_x (u^2 + w^2) = K_F \ C_x \ V^2 \cos \beta_s$$
$$F_{y_v} = K_F \ C_y (u^2 + v^2) = K_F \ C_y \ V^2 (\cos^2 \beta_s \cos^2 \alpha + \sin^2 \beta_s)$$
$$F_{z_v} = K_F \ C_z (u^2 + w^2) = K_F \ C_z \ V^2 \cos^2 \beta_s$$
$$M_p = K_M \ C_m (u^2 + w^2) = K_M \ C_m \ V^2 \cos^2 \beta_s$$
$$M_y = K_M \ C_n (u^2 + v^2) = K_M \ C_n \ V^2 (\cos^2 \beta_s \cos^2 \alpha + \sin^2 \beta_s)$$
$$M_R = K_M \ C_l (V^2 + w^2) = K_M \ C_l \ V^2 (\cos^2 \beta_s \sin^2 \alpha + \sin^2 \beta_s)$$

2.1.2. Jet Engine Parameters. - Methods for obtaining the forces and moments on the jet engine are given in this paragraph. In order to determine the total forces and moments on the vehicle, it is necessary to obtain the magnitudes of the forces (and moments) on the jet engine, transfer these to the vehicle body axes, and add them to the outer frame parameters presented in Paragraph 2.1.1.

The three components of the velocity in the body axis system are u, v, and w. The engine angle of attack is given by:

$$\sin \alpha_e = \frac{W_e}{V}$$

where W_e is the component of the velocity along the engine Z_e axis. The transformation matrix from the vehicle to the engine, using the gimbal angles θ_{jv} (pitch) and ϕ_{jv} (roll), is given in Figure 2-2.

The velocity component, W_e, is given by:

$$W_e = u \cos \phi_{jv} \sin \theta_{jv} - v \sin \phi_{jv} + w \cos \phi_{jv} \cos \theta_{jv}$$

and

$$\begin{aligned} \text{Sin } \alpha_e &= \frac{u}{V} \cos \phi_{jv} \sin \theta_{jv} - \frac{u}{V} \sin \phi_{jv} + \frac{w}{V} \cos \phi_{jv} \cos \theta_{jv} \\ &= \cos \alpha \cos \beta_s \cos \phi_{jv} \sin \theta_{jv} - \sin \beta_s \sin \phi_{jv} + \\ &\qquad \sin \alpha \cos \beta_s \cos \phi_{jv} \cos \theta_{jv} \end{aligned}$$

Once the engine angle of attack, α_e, is determined the magnitude of the drag, D_e, and moment, M_e, can be determined. The following equations are based on wind tunnel tests of an 0.3 scale model of the LLRV jet engine. The equation for X_{cp} was obtained by a fit of the center-of-pressure data from the wind tunnel tests; $\dot{w}$ is based on average airflow values given in Reference 1.

The moment is given by:

$$M_e = \frac{\dot{w}V}{g} \cos \alpha_e \quad X_{cp}$$

and the drag by:

$$D_e = \frac{\dot{w}V}{g} \cos \alpha_e + 11.7q$$

Where $X_{cp} = 2.2855\ (1.0 + 0.5109 \sin \alpha_e) \sim \text{ft}$

$\dot{w} = 24.5 + 36.6\ (T/1000) - 3.303\ (T/1000)^2 \sim \text{lb/sec}$

T = Jet engine Thrust $\sim$ lb

α_e = Engine angle-of-attack

q = Dynamic pressure

The forces and moments in the engine axis system are defined in Figure 2-4.

Once the magnitudes are known, it is a simple matter to determine the components of the engine drag in the body axis system. These are, since the drag is along the velocity vector, given by:

$$F_{x_e} = D_e \frac{u}{V} = D_e \cos \alpha \cos \beta_S$$

$$F_{y_e} = D_e \frac{v}{V} = D_e \sin \beta_S$$

$$F_{z_e} = D_e \frac{w}{V} = D_e \sin \alpha \cos \beta_S$$

The engine moment is more difficult to transfer to the vehicle axes. The simplest way is to represent the moment by a vector which is defined by:

$$\underline{M}_e = (M_e)\ (\bar{A}_e \times \frac{\bar{V}}{|\underline{V}|})$$

where A_e is a unit vector along the engine Z_e axis given by:

$$A_e = \cos \phi_{jv} \sin \theta_{jv}\ \underline{i} - \sin \phi_{jv}\ \underline{j} + \cos \phi_{jv} \cos \theta_{jv}\ \underline{k}$$

and $\frac{V}{|\underline{V}|}$ is a unit vector given by:

$$\frac{V}{|\underline{V}|} = \frac{u}{V}\ \underline{i} + \frac{v}{V}\ \underline{j} + \frac{w}{V}\ \underline{k}$$

$$= \cos \alpha \cos \beta_s\ \underline{i} + \sin \beta_s\ \underline{j} + \cos \beta_s \sin \alpha\ \underline{k}$$

After performing the necessary algebra the vector M can be written as:

$$\begin{aligned} M_e = \left[M_e\right] \Big\{ & - (\cos \phi_{jv} \cos \theta_{jv} \sin \beta_s + \cos \beta_s \sin \alpha \sin \phi_{jv})\ \underline{i} \\ & + (\cos \beta_s \cos \phi_{jv} (\cos \alpha \cos \theta_{jv} - \sin \alpha \sin \theta_{jv})\ \underline{j} \\ & + (\cos \beta_s \cos \alpha \sin \phi_{jv} + \sin \beta_s \cos \phi_{jv} \sin \theta_{jv})\ \underline{k} \Big\} \end{aligned}$$

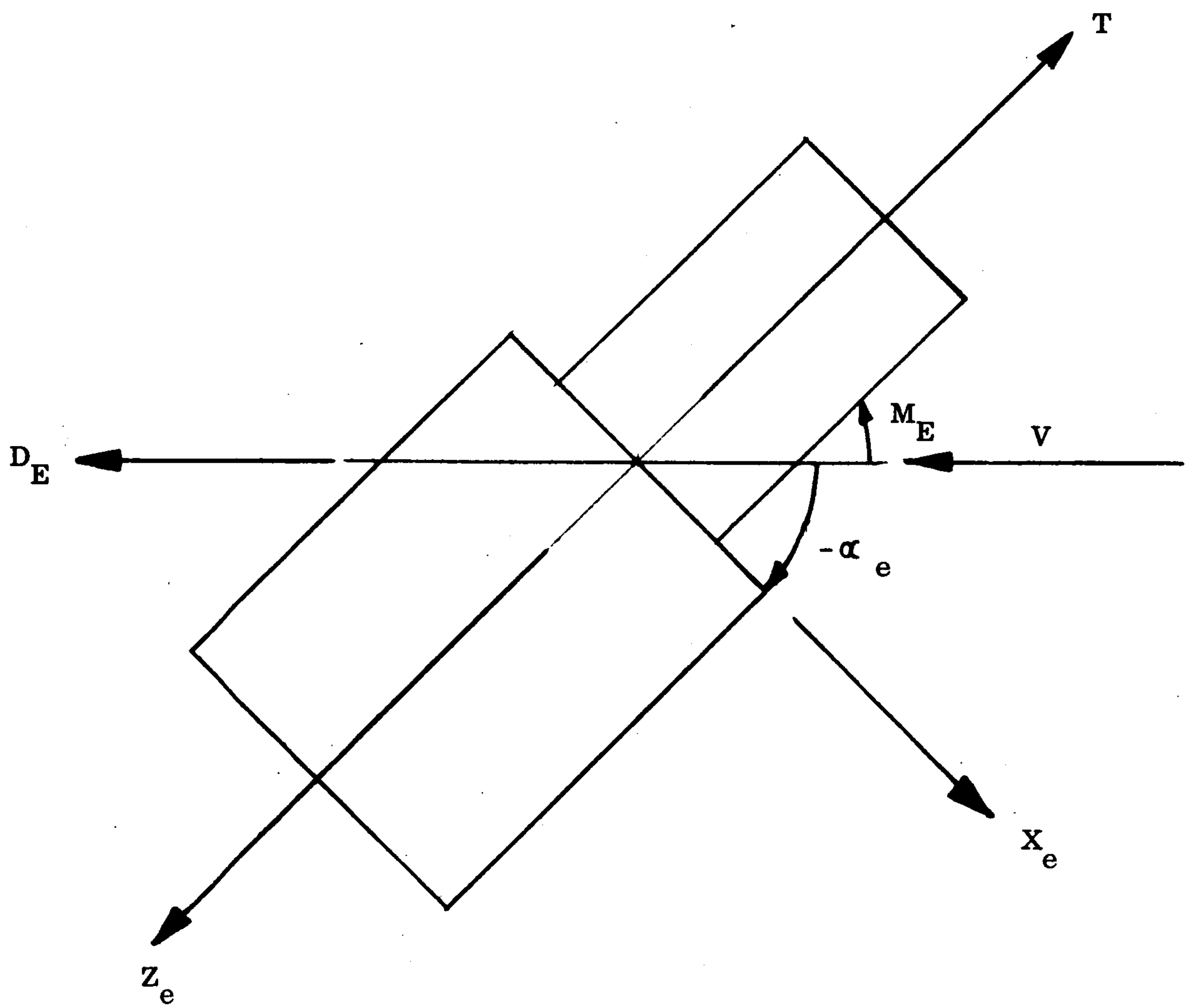


Figure 2-4. Engine Force and Moment

The three components of the vector M_e are the contributions of the engine moment to the roll, pitch and yaw moments around the body axes. That is:

$$\text{(Pitch)} \quad P_e = M_e (\cos \beta_s \cos \phi_{jv}) (\cos \alpha \cos \theta_{jv} - \sin \alpha \sin \theta_{jv})$$
$$\text{(Yaw)} \quad Y_e = M_e (\cos \beta_s \cos \alpha \sin \phi_{jv} + \sin \beta_s \cos \phi_{jv} \sin \theta_{jv})$$
$$\text{(Roll)} \quad R_e = -M_e (\cos \phi_{jv} \cos \theta_{jv} \sin \beta_s + \cos \beta_s \sin \alpha \sin \theta_{jv})$$

2.1.3. Total Vehicle Parameters..- Methods for determining the forces and moments of the outer frame and jet engine separately have been given in Paragraphs 2.1.1 and 2.1.2. These parameters are both referred to the body axis system and therefore can be added directly to obtain the total aerodynamic parameters on the vehicle.

2.2. JET ENGINE PROPULSION SYSTEM

The primary propulsion system in the LLRV is a G.E. CF700-2V turbofan jet engine. The engine to be used in the first vehicle is designated serial number 239001.

2.2.1. Basic Engine Performance. - The CF700-2V SLS (sea level, standard day) performance is summarized in Table 2-2. Temperature corrections to the SLS Thrust are given in Figure 2-5. Figure 2-6 presents the variation of thrust versus altitude.

TABLE 2-2.

CF700-2V SLS PERFORMANCE

REFERENCE		TAKEOFF POWER		MAXIMUM CONTINUOUS (MILITARY)	
		Fg	SFC	Fg	SFC
2	Guaranteed Minimum	4200	0.7	4000	0.69
3	Acceptance Test Data	4320 (@1310°EGT)	0.66	4105	0.65
1	Estimated Average	4300	0.67	4170	0.66

2.2.2.. Installation Losses. - Calculated installation losses amount to 0.93 percent; 0.83 percent is contributed by the inlet and 0.1 percent by power extraction. In order to simplify calculations it was assumed that the gross thrust would be reduced 1.0 percent due to installation losses. There is no change in the fuel flow.

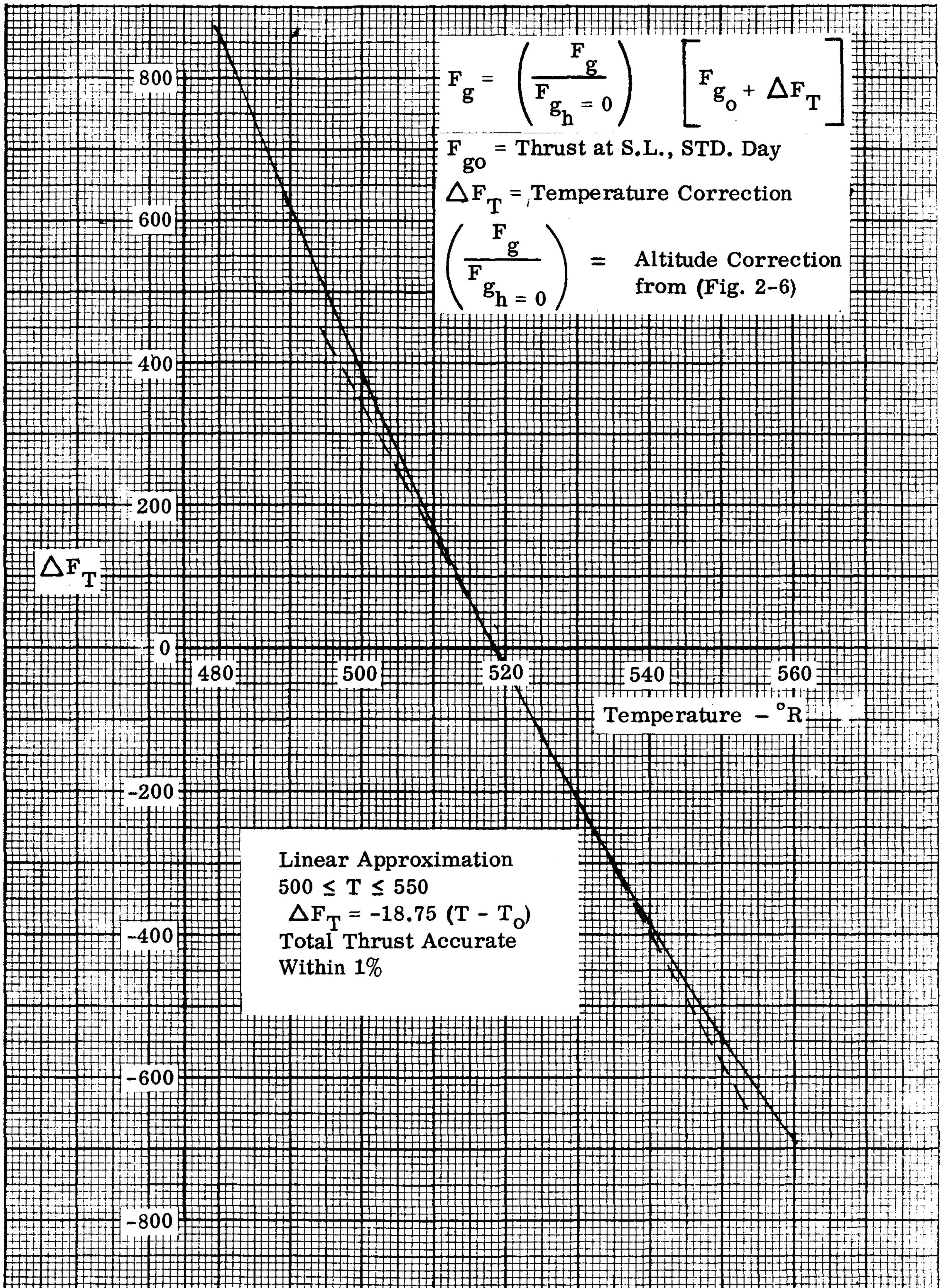


Figure 3 CF700-2V Takeoff Thrust – Temperature Correction

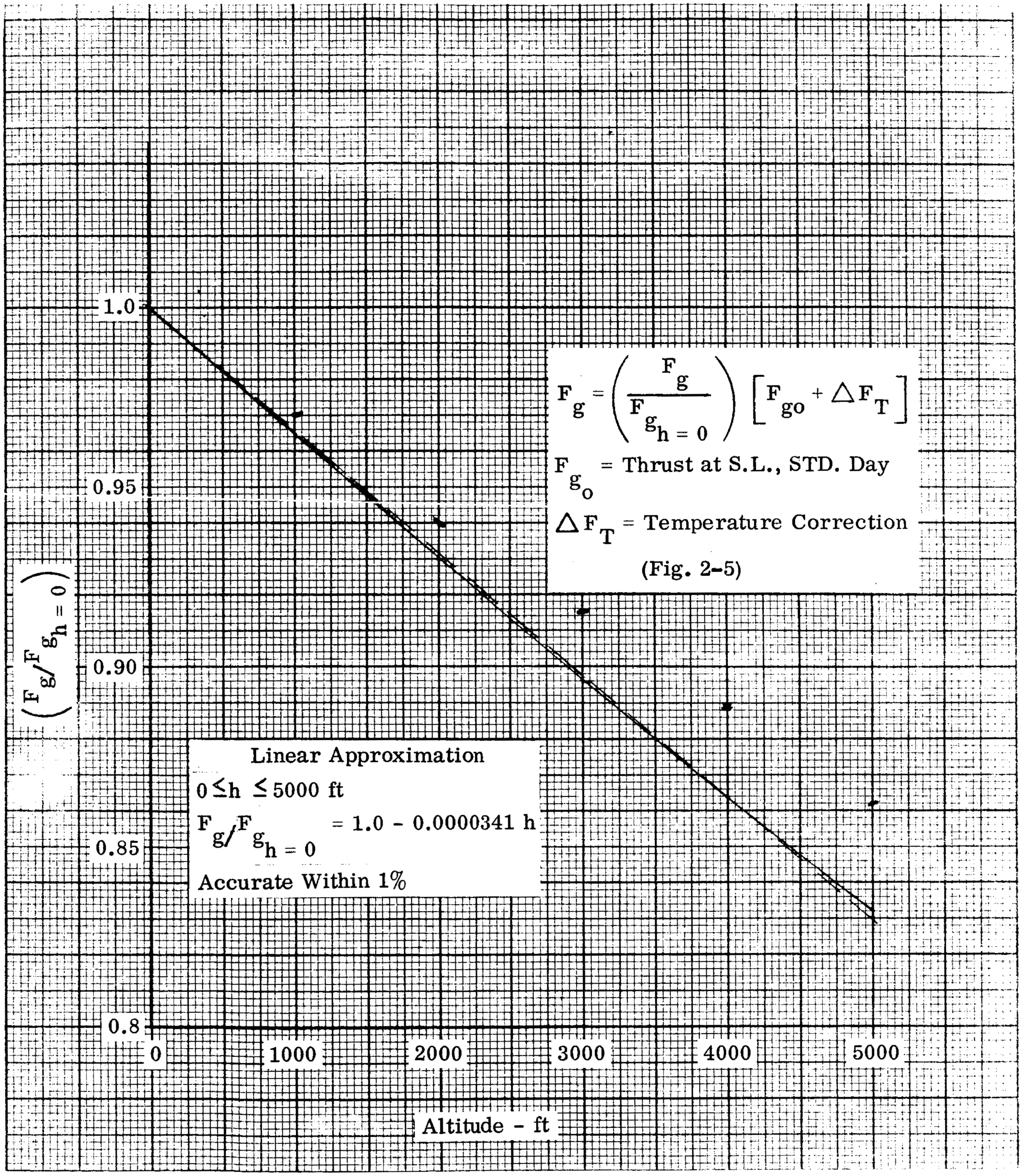


Figure 2 CF700-2V Takeoff Thrust – Altitude Correction

2.2.3. Effects of Lateral and Vertical Velocity. - Wind tunnel tests of the LLRV, along with experience gained from other types of VTO aircraft, indicated that the following effects could be expected due lateral and vertical velocity variations.

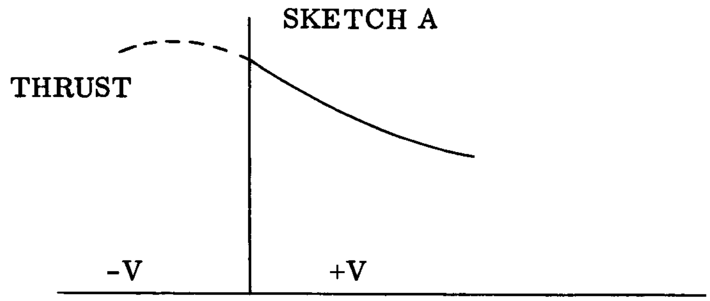
(1) The thrust variations due to velocity are at most 5 percent of the takeoff thrust.

(2) The loss is most severe during vertical ascent, when the engine angle of attack is zero. The increment in thrust due to velocity, ΔF_V, is given by

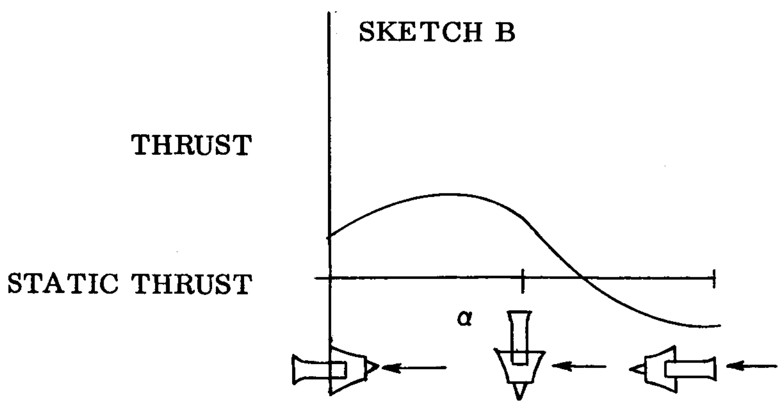
$$\Delta F_V = -3.12 V \quad \text{lb}$$

(3) The variation is beneficial (i.e., would increase the thrust) during level flight and descent operations. This is shown in the following sketches:

Sketch A shows the variation of jet engine thrust versus velocity (+V is ascending, -V is descending). The dashed portion of the curve is seldom shown in engine performance manuals but is typical for this case.



Sketch B shows the variation of thrust with angle-of-attack at constant velocity. It can be seen that at some point when the engine axis is at an angle to the airflow, the thrust increases over the static value. The crossover point varies with velocity and engine airflow, but is always less than 90 degrees for the LLRV flight conditions.



Based on these data, a decision was made to include thrust variations due to velocity only during computation of the vertical ascent to altitude. Performance calculations during translation are greatly simplified by this approach.

2.2.4. Engine Performance. - The CF700-2V installed performance characteristics are summarized in Figures 2-7 through 2-10.

Variations of the net thrust with ambient temperature and altitude, Figures 2-7 and 2-8, were computed from

$$F_N = (F_N/F_g) \left(\frac{F_g}{F/gh = 0}\right) (F_{go} + \Delta F_T)$$

where:

F_N/F_g = 0.99 (Installation losses)

$F_g/F_{gh=0}$ - Figure 2-6 (Altitude correction)

F_{go} - SLS Gross Thrust (No losses)

ΔF_T - Figure 2-5 (Temperature Correction)

F_N - Net Thrust

The specific fuel consumption, Figure 2-9, is based on the net thrust and guaranteed values of specific fuel consumption. The fuel flows shown in Figure 2-10 correspond to the estimated average thrust values.

2.3. ROCKET SYSTEM PERFORMANCE.

2.3.1. Lift Rockets. - The lift rocket system performance which was used in the performance calculations is given in Figure 2-11.

During normal operation with two lift rockets,

$$\dot{W}_R = 0.7 + 0.00718\ T_R \quad \sim \quad \text{lb/sec}$$

T_R = Thrust of two lift rockets

During an emergency situation requiring all eight lift rockets, the fuel flow would be:

$$\dot{W}_{ER} = 2.8 + 0.00718\ T_{ER}$$

T_{ER} = Thrust of eight lift rockets.

The maximum rocket thrust which can be provided during an emergency is 3752 lb.

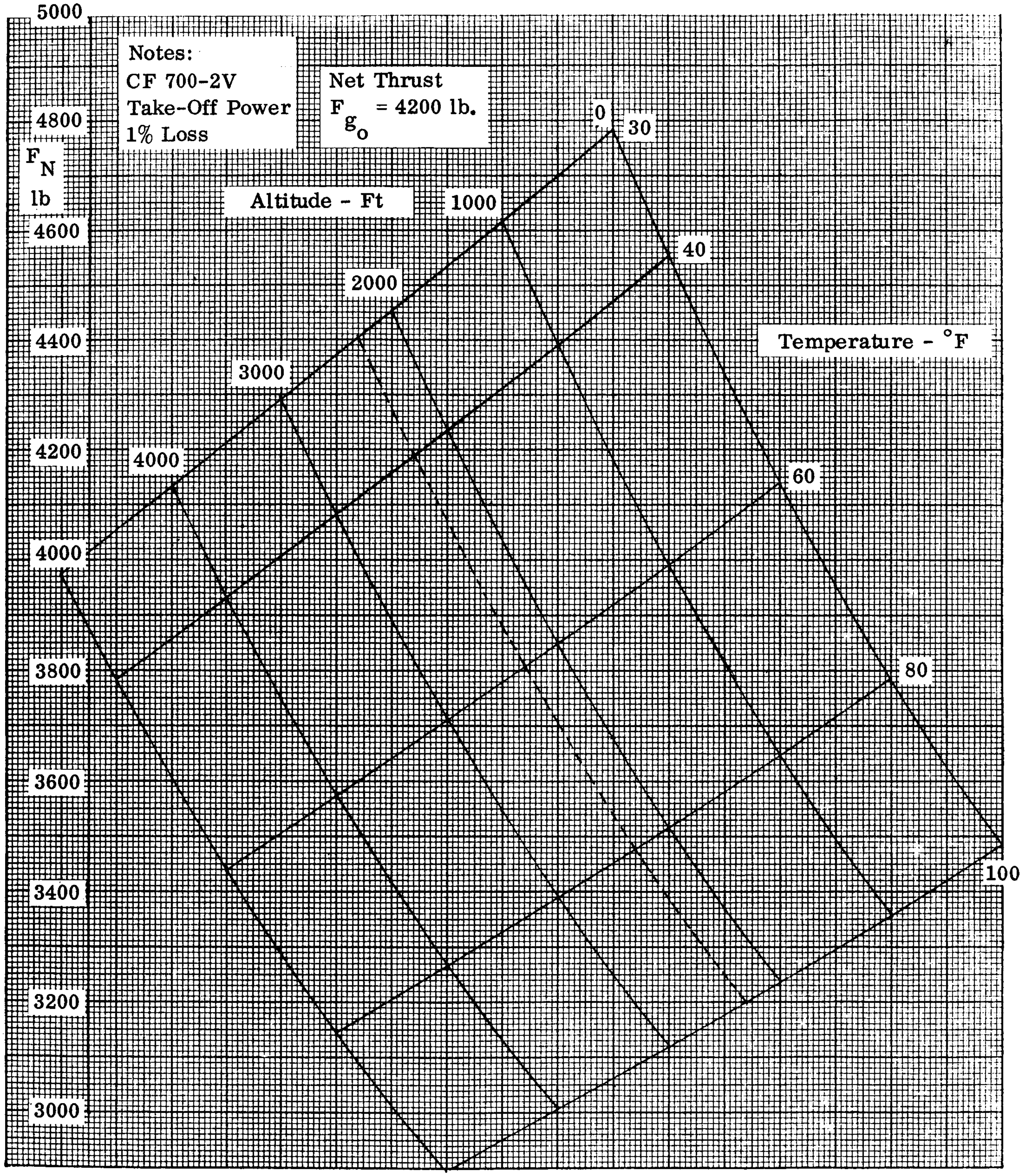


Figure 2-7. Variations of Net Thrust with Ambient Temperature – Guaranteed Minimum

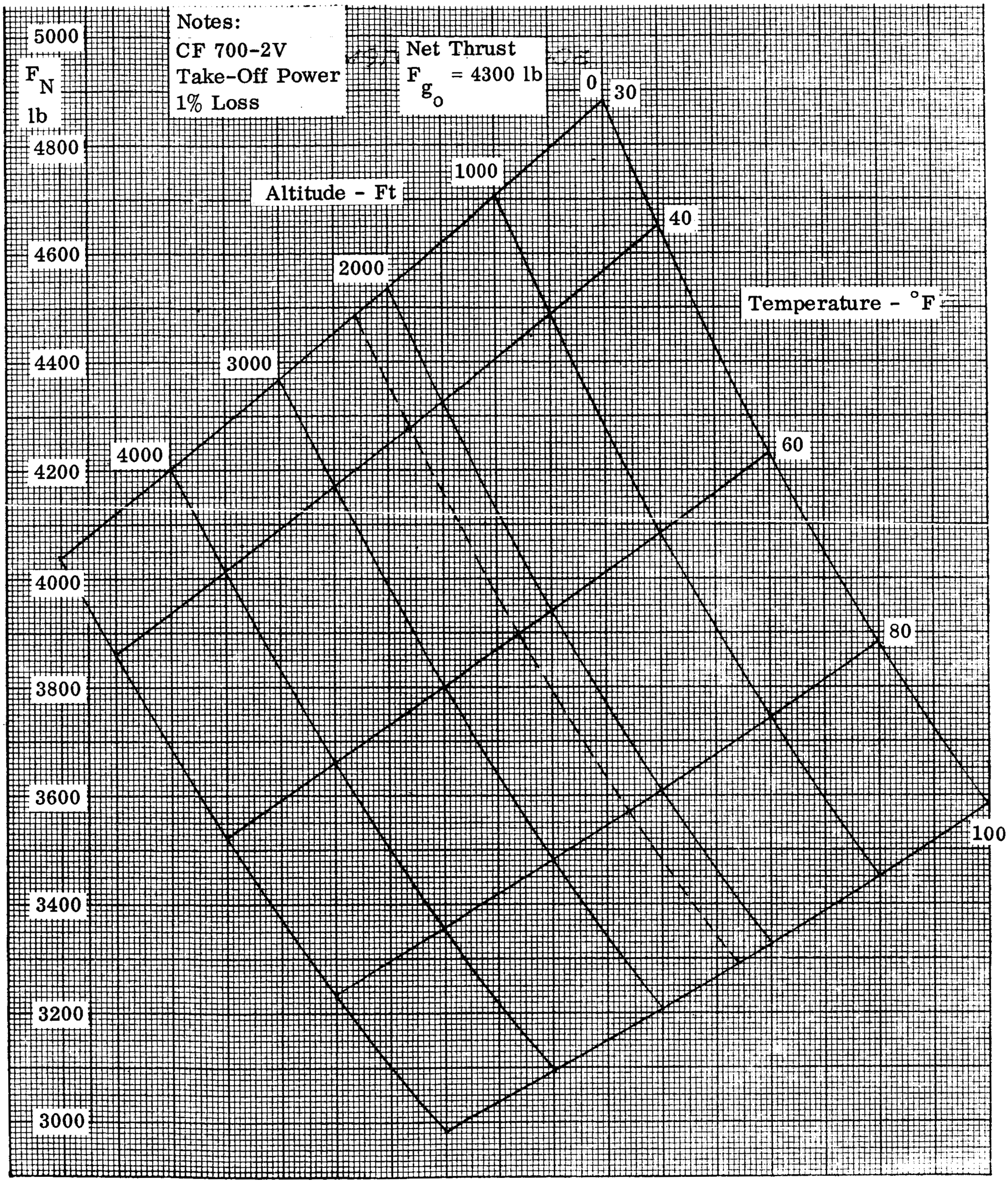


Figure 1 Variation of Net Thrust with Ambient Temperature – Estimated Average

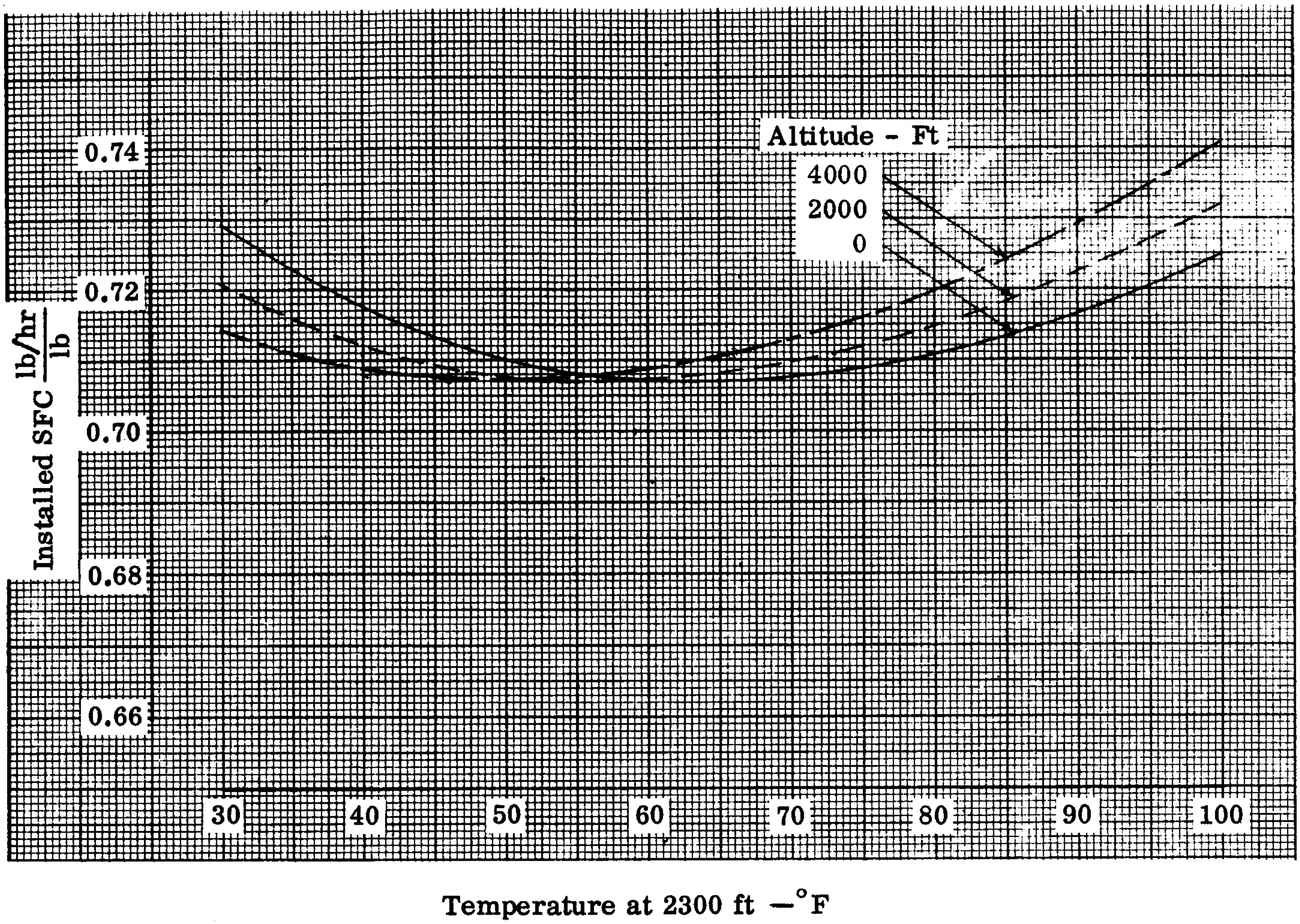


Figure 2-9. Installed Specific Fuel Consumption

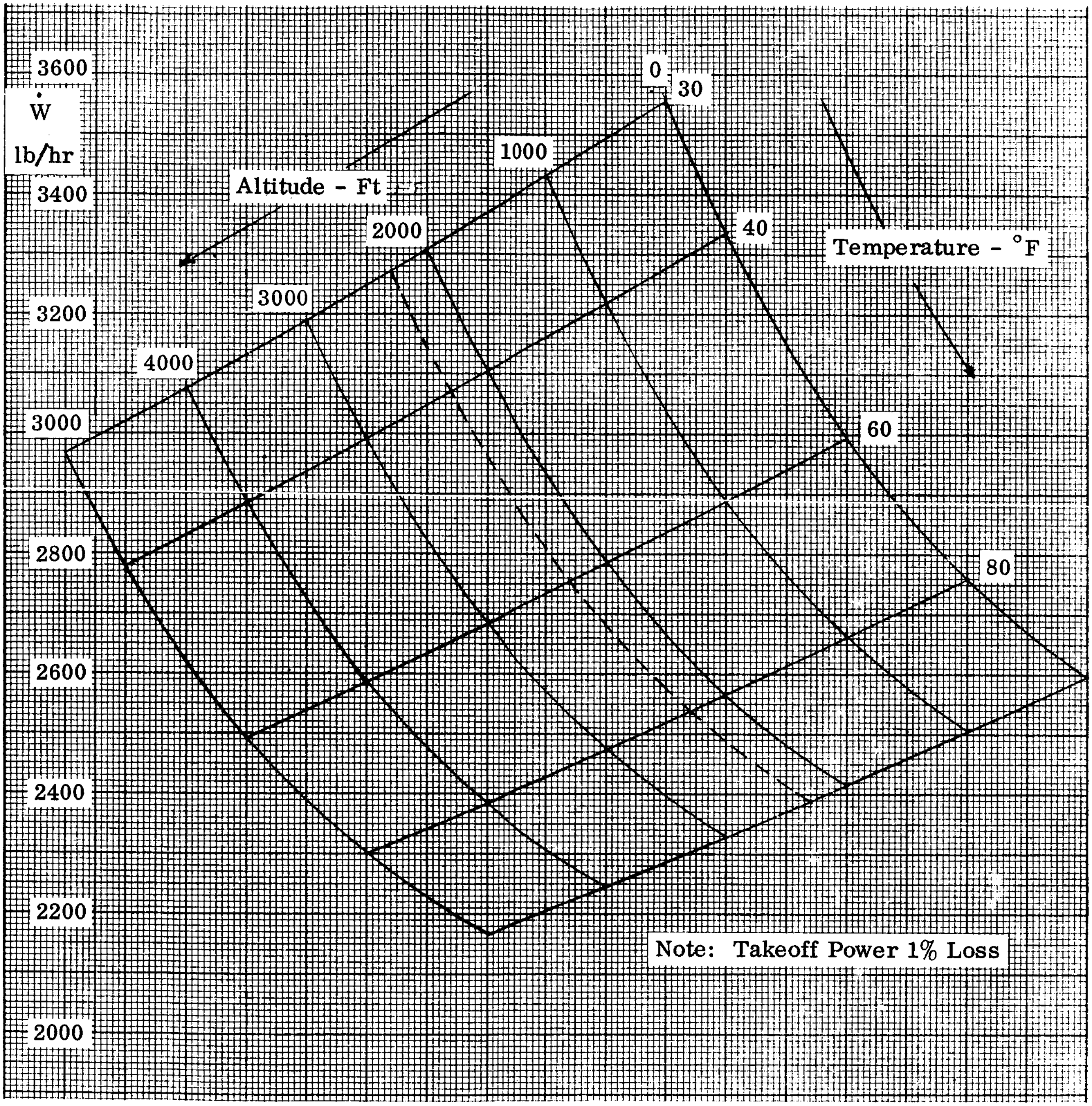


Figure 2-10. Fuel Flow – Estimated Average

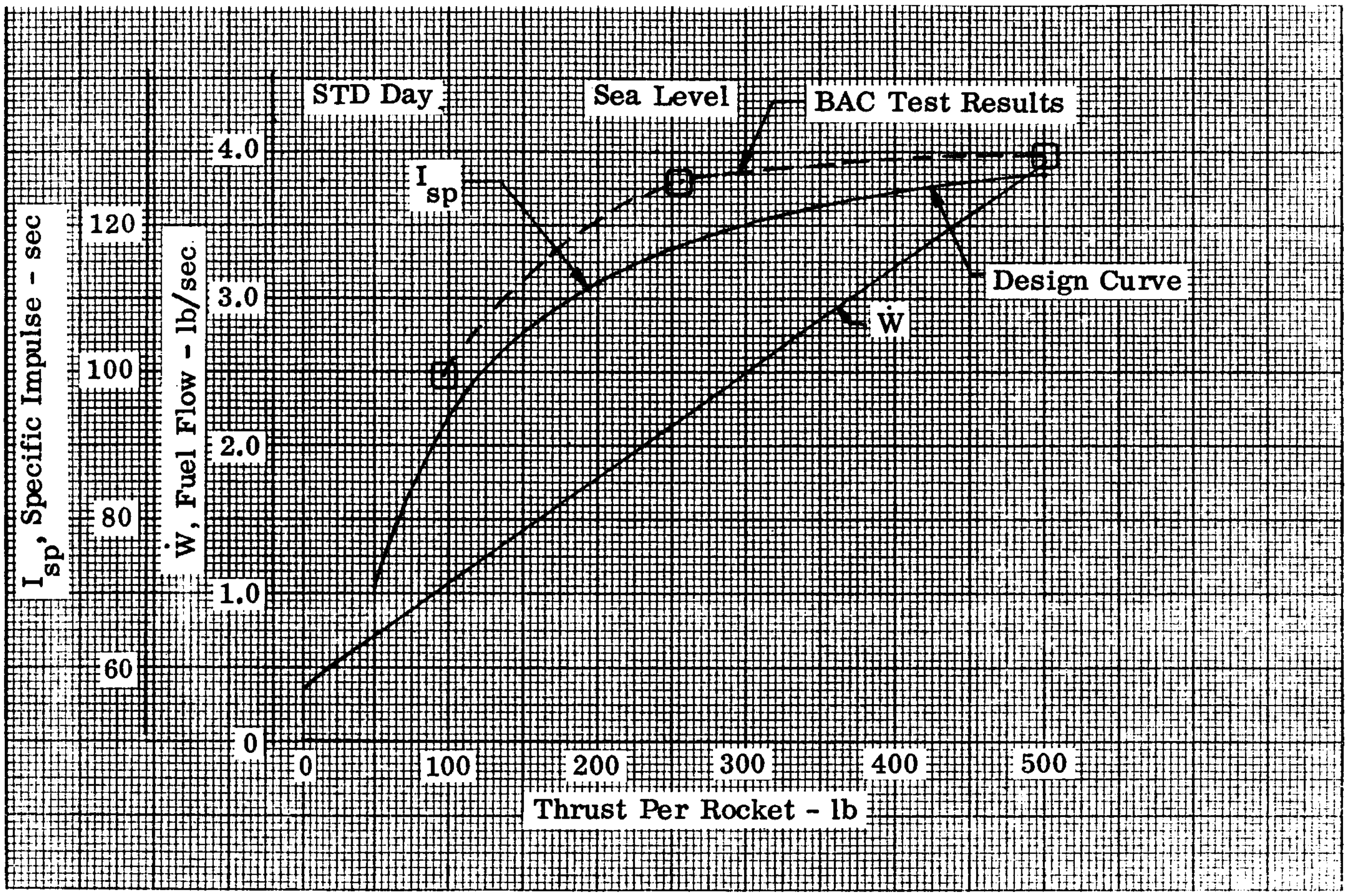


Figure 2-11. Lift Rocket Performance

2.3.2. Attitude Control Rockets. - The attitude control rockets have a specific impulse of 60 to 120 sec; depending on the duty cycle. For use in vehicle performance calculations, a specific impulse of 100 sec is suggested as a reasonable average.

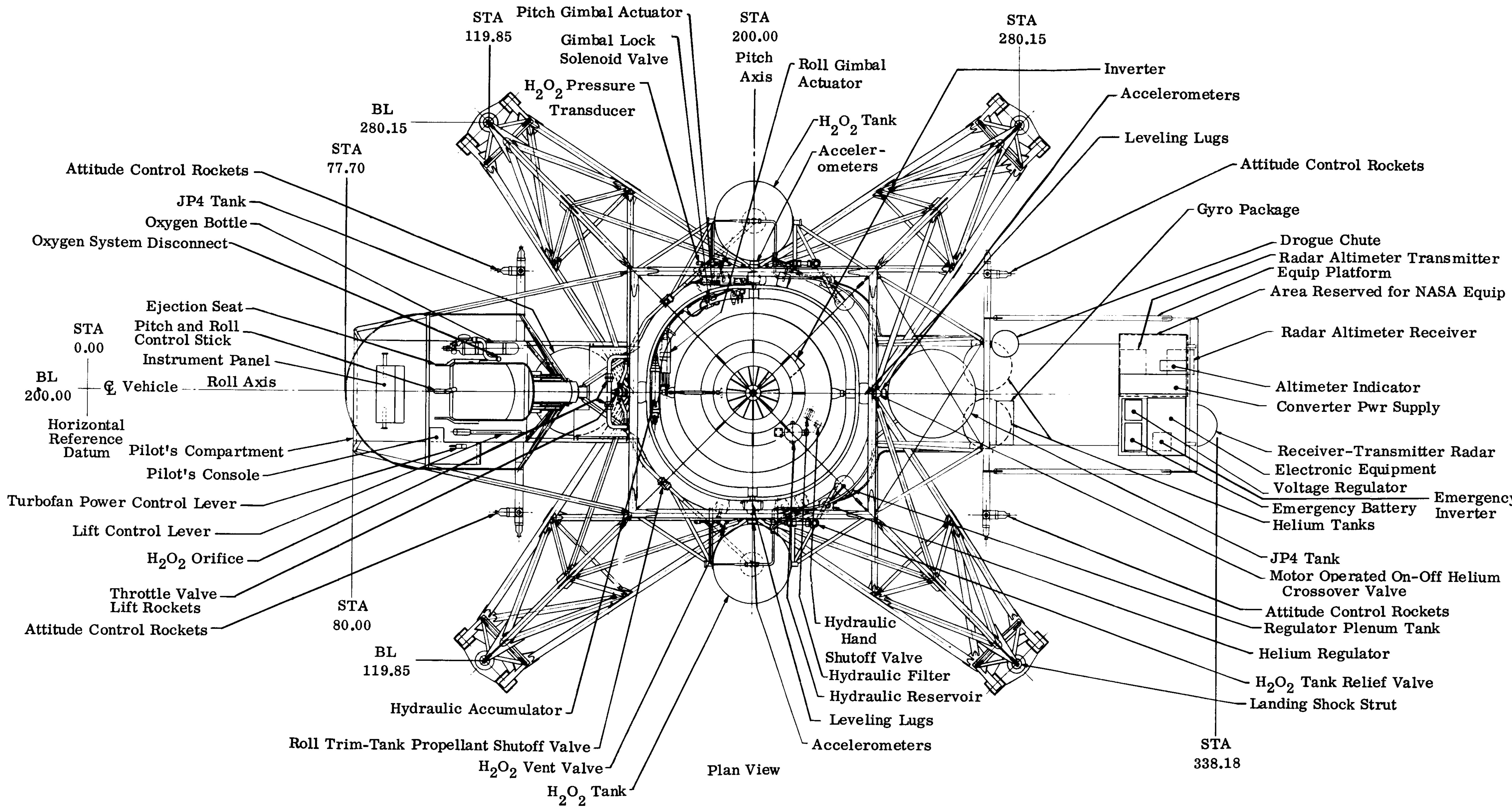
2.4. DIMENSIONAL DATA AND WEIGHTS.

The general arrangement of the LLRV is given in Figure 2-12. Table 2-3 gives the weight data used for performance computations.

TABLE 2-3.

WEIGHT STATEMENT

	LIGHT PILOT	HEAVY PILOT
Empty Weight	2466	2466
Pilot	150	200
JP4	400	400
H_2O_2	600	600
Gross Weight	3616	3666



STA 119.85
Pitch Gimbal Actuator
Gimbal Lock Solenoid Valve
H_2O_2 Pressure Transducer
STA 200.00
Pitch Axis
Roll Gimbal Actuator
H_2O_2 Tank
Acceler-ometers
STA 280.15
Inverter
Accelerometers
Leveling Lugs
Attitude Control Rockets
Gyro Package
Drogue Chute
Radar Altimeter Transmitter
Equip Platform
Area Reserved for NASA Equip
Radar Altimeter Receiver
Altimeter Indicator
Converter Pwr Supply
Receiver-Transmitter Radar
Electronic Equipment
Voltage Regulator
Emergency Battery
Emergency Inverter
Helium Tanks
JP4 Tank
Motor Operated On-Off Helium Crossover Valve
Attitude Control Rockets
Regulator Plenum Tank
Helium Regulator
H_2O_2 Tank Relief Valve
Landing Shock Strut
STA 338.18
BL 280.15
STA 77.70
Attitude Control Rockets
JP4 Tank
Oxygen Bottle
Oxygen System Disconnect
Ejection Seat
Pitch and Roll Control Stick
Instrument Panel
STA 0.00
BL 200.00
℄ Vehicle
Roll Axis
Horizontal Reference Datum
Pilot's Compartment
Pilot's Console
Turbofan Power Control Lever
Lift Control Lever
H_2O_2 Orifice
Throttle Valve Lift Rockets
Attitude Control Rockets
STA 80.00
BL 119.85
Hydraulic Accumulator
Roll Trim-Tank Propellant Shutoff Valve
H_2O_2 Vent Valve
H_2O_2 Tank
Hydraulic Hand Shutoff Valve
Hydraulic Filter
Hydraulic Reservoir
Leveling Lugs
Accelerometers

Plan View

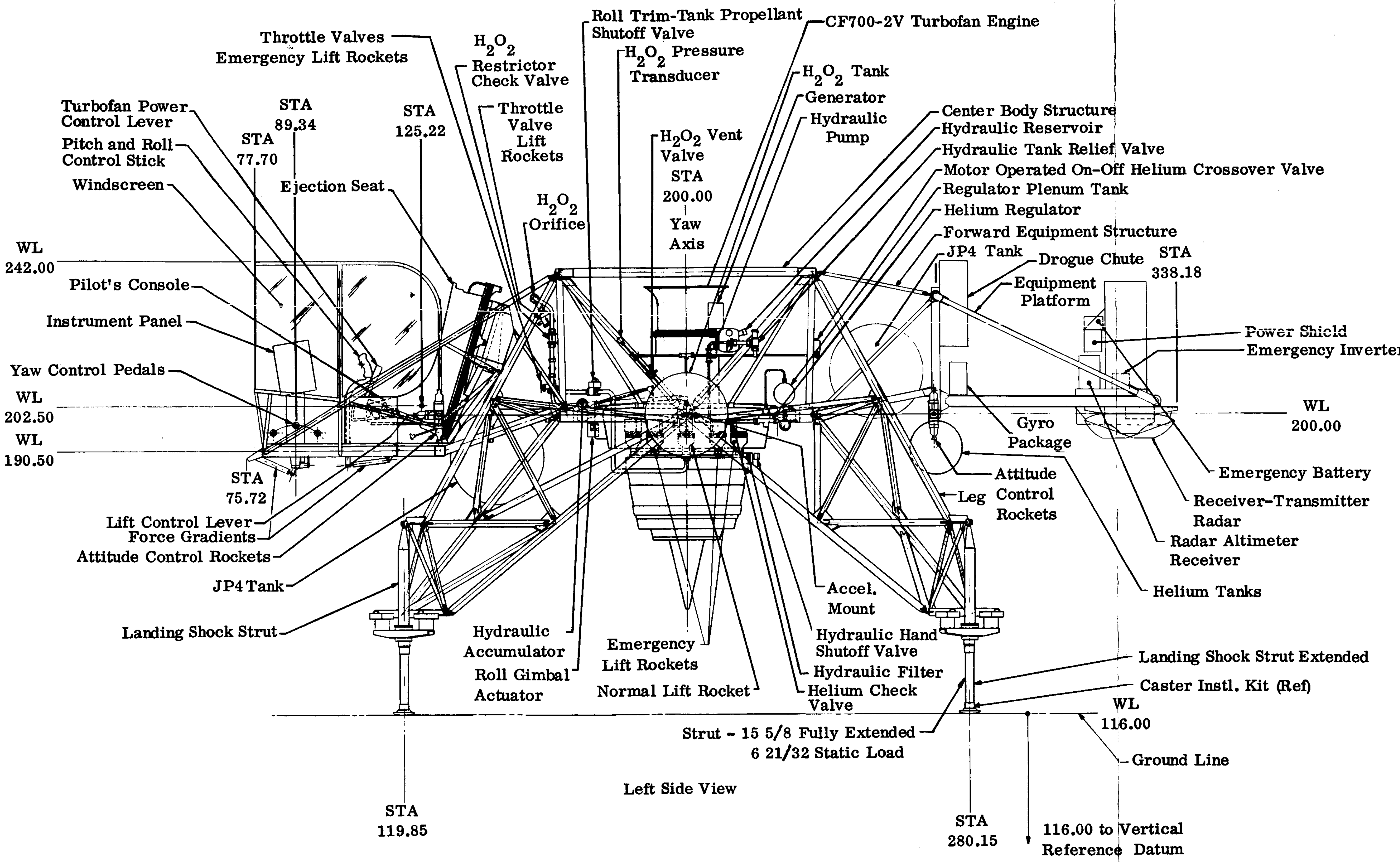


Figure 2-12. LLRV General Arrangement

SECTION III
PERFORMANCE

3.1. GENERAL.

This Section presents the equations necessary to determine the fuel requirements for various LLRV missions. Simplified linear expressions are given, along with charts to facilitate their evaluation. These methods and equations are designed to produce slightly conservative estimates with a minimum of time and effort. The expressions presented have been checked against more accurate results, and, in some cases, against digital computer solutions to insure their validity.

Since the thrust and fuel flow are a strong function of temperature, a number of the curves have been plotted against the temperature at 2300 feet (i.e., elevation of NASA - FRC) this will facilitate computations during operation of the vehicle at this site.

3.2. MISSION FUEL REQUIREMENTS.

3.2.1. Jet Fuel Required. - The jet fuel required for a 10-minute flight with no lunar simulation is shown in Figure 3-1 as a function of the takeoff weight and the SFC. This data can be represented by the equation:

$$W_j = (0.2644 \times 10^{-3}) \quad t \quad (SFC) \quad W_i$$

where t ~ Total flight endurance.

In order to provide a conservative estimate of the JP4 required for the 10-minute flight, it was assumed that no H_2O_2 was consumed. That is, the only weight variation was due to the usage of JP4. This assumption gives JP4 fuel weights which are conservative by about two percent.

The maximum endurance of the LLRV as a function of takeoff weight and SFC is shown in Figure 3-2. The assumptions of the previous paragraphs were used to find the total flight time for 400 pounds of jet engine fuel.

3.2.2. Rocket Fuel Required. - The hydrogen peroxide used by the lift rockets during the lunar simulation is highly dependent on the type of mission and the pilots handling of the lift rocket throttle. The effects of these parameters can only be studied realistically through

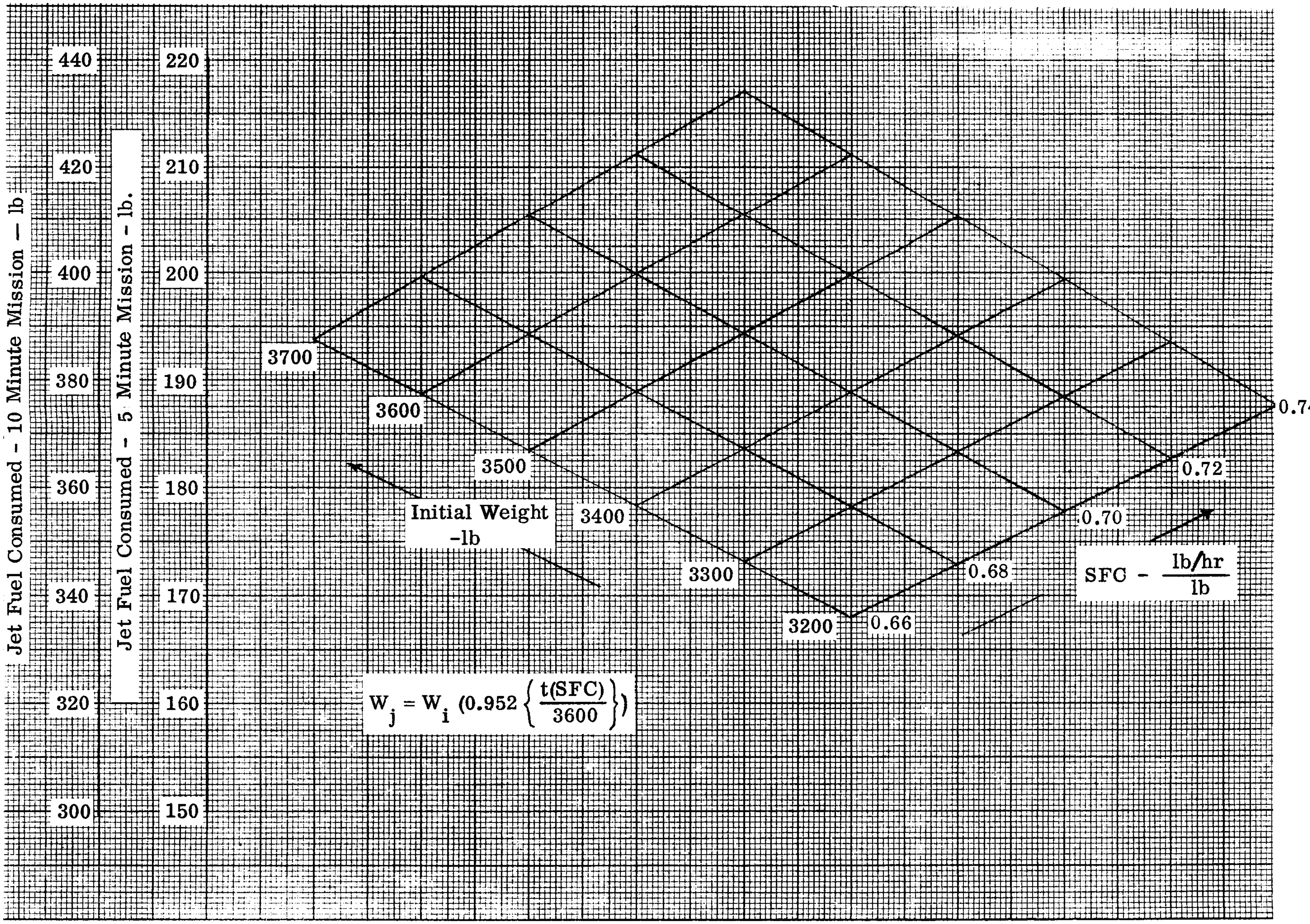


Figure 3-1. Jet Fuel Consumption

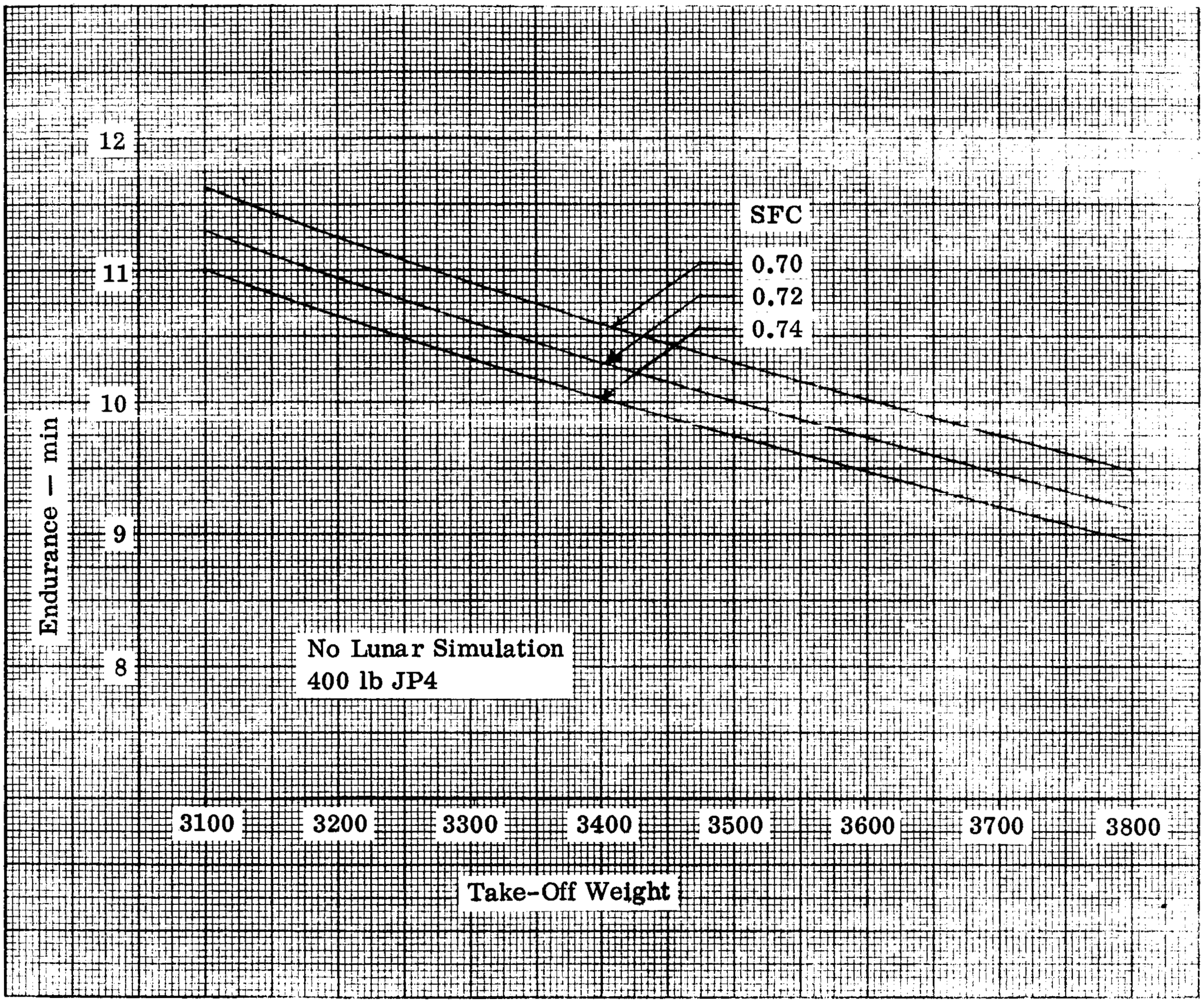


Figure 3-2. Endurance versus Takeoff Weight

an analog type simulation with a pilot in the loop. However, a reasonable estimate of the total fuel used and the effects of vehicle weight and K_R can be made by assuming that equilibrium flight conditions (i.e., total thrust equals weight) hold during the simulation period. If K_R is the fraction of the vehicle weight, W, supported by the lift rockets then:

$$K_R W = T_R = \text{lift rocket thrust.}$$
$$(1-K_R)W = T_j = \text{jet engine thrust}$$

and the instantaneous fuel flow is:

$$\frac{dw}{dt} = -\frac{(1-K_R)\ W\ (SFC)}{3600} - (0.7 + 0.00718\ (K_R W)) \sim \quad \text{lb/sec}$$

By letting the SFC take on a constant value (0.7 was used), this equation can be integrated to obtain the total fuel (JP4 plus H_2O_2) used. The JP4 used during the simulation was subtracted from the total to find the lift rocket fuel. This latter weight for a two-minute simulation is given in Figure 3-3. This data can be represented by the following equation:

$$W_R = \left[82 + (0.001 + 0.78\ K_R)\ W_s\right] \ t/120$$

where

W_R = weight of H_2O_2 consumed by the lift rockets

t = time of simulation

W_s = vehicle weight at the start of simulation

The total hydrogen peroxide required is the amount needed by the lift rockets during simulation plus the amount used by the attitude controls during the entire flight. There can be a substantial variation in the useage of the attitude controls from flight to flight. Simulator studies have indicated flow rates from six to twenty pounds per minute. Until final information is obtained from the planned FRC simulation, a weight of 150 pounds is suggested as a conservative estimate of the amount of H_2O_2 which should be allotted for attitude control.

The weight and drag computer, which sends signals to the automatic jet throttle controls during lunar simulations, has a feature which permits the jet engine to provide a constant thrust increment in addition to the 5/6 W required for lunar simulation. This increment can be used to simulate a constant lift rocket thrust. When this is done the actual lift rockets can be operated at a lower level and throttled to proxide "vernier" descent

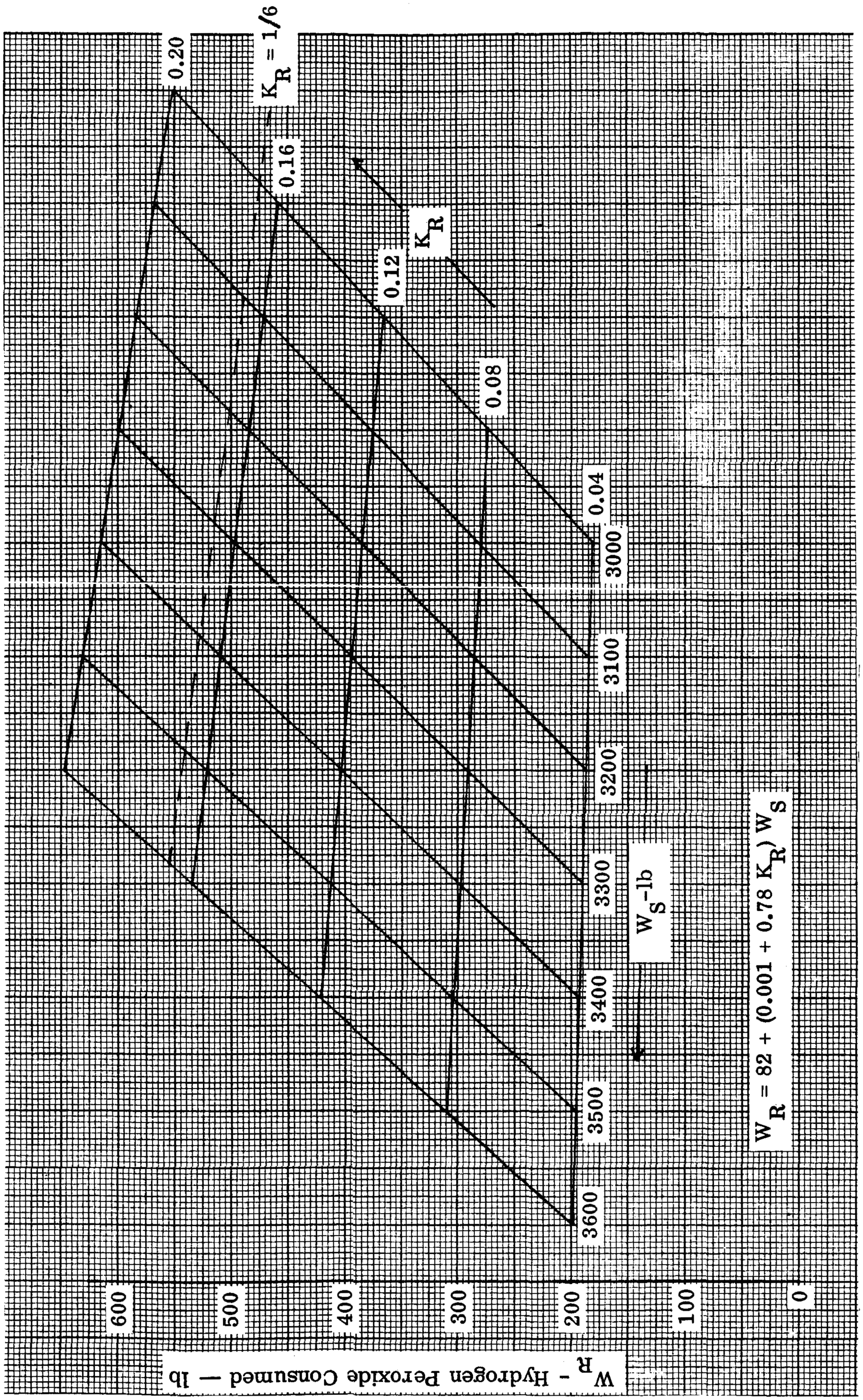


Figure 3-3. Lift Rocket Fuel Consumed During Two-Minute Simulation

control. In this way the H_2O_2 requirements can be considerably reduced. A good estimate of the H_2O_2 requirements can be obtained by computing an average K_R and using Figure 3-3.

3.3. TAKEOFF AND CLIMB.

Figure 3-4 shows the maximum net thrust at 2300 and 4000 feet. It is plotted against the temperature at 2300 feet. The normal temperature gradient is used to determine the 4000-foot case. The maximum gross takeoff weight, $W_{T.O.}$, is specified (Reference 5) as 0.955 of the net thrust and is shown in Figure 3-5. It can be seen by comparing Figures 3-4 and 3-5 that there is a small but definite excess thrust $(F_N - W_{T.O.})$ at 4000 feet for all takeoff weights. This assures that the LLRV will be able to maintain a positive rate of climb to 4000 feet. The jet fuel required and time to climb from 2300 feet to 4000 feet are shown in Figure 3-6 versus the thrust-to-weight ratio at takeoff. The curves shown were computed using the following parameters:

ALTITUDE	VELOCITY	THRUST	SFC
2300	0	3839	0.7075
	100	3527	0.7075
4000	0	3603	0.7075
	100	3291	0.7075

$C_D = 1.83$

$$W_{T.O.} = \frac{3839}{(T/W)_{T.O.}}$$

Attitude control fuel flows of zero and 15 pounds per minute were examined along with T/W ratios less than the specified value of 1.047 $\left(\frac{1}{0.955}\right)$.

3.4. TRANSLATIONAL FLIGHT.

3.4.1. Gimbals Fixed. - The LLRV is designed to translate by tilting the entire vehicle. This produces a component of thrust along the flight path which accelerates the vehicle until the thrust component is equal to the drag. The jet engine is limited to a maximum tilt angle of 15 degrees by the design of the lubrication system. This is sufficient to accelerate the vehicle well beyond the design speed of 60 feet per second. Increases in thrust or fuel flow during translation over the values for hover are insignificant; therefore, the procedures already outlined for the calculation of fuel consumed during hover are applicable.

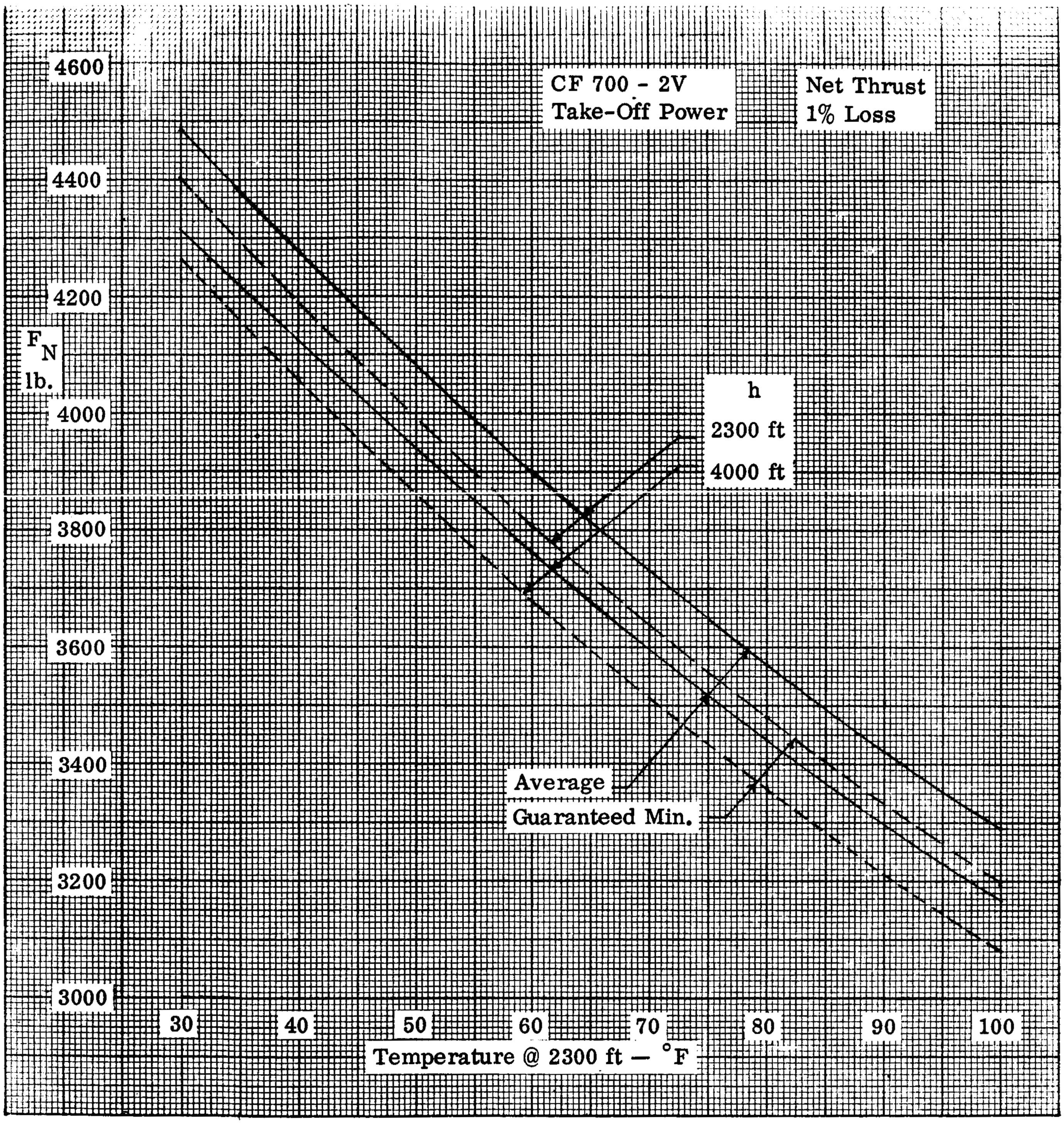


Figure 3-4. Maximum Net Jet Engine Thrust

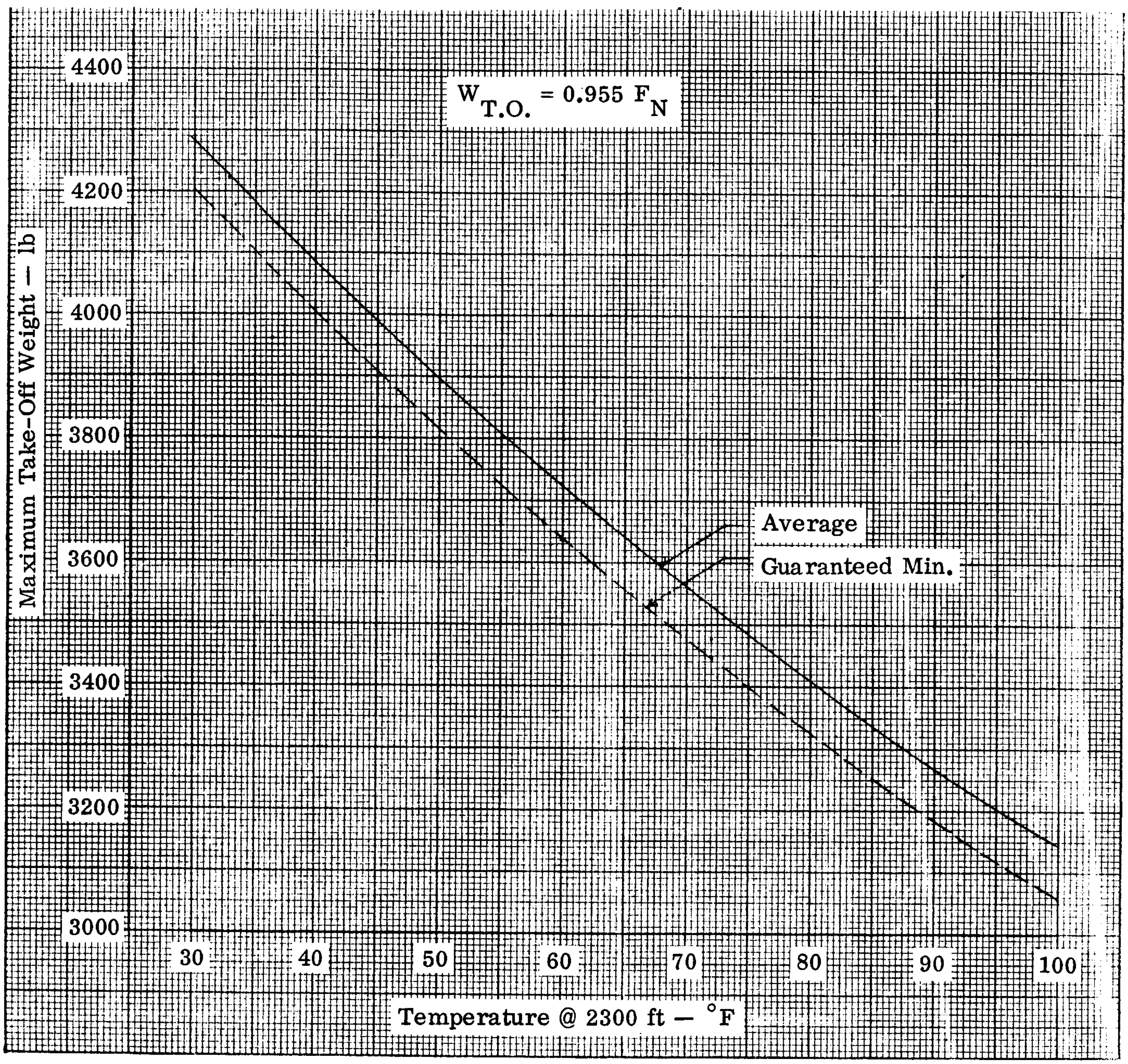


Figure 3-5. Maximum Gross Takeoff Weight

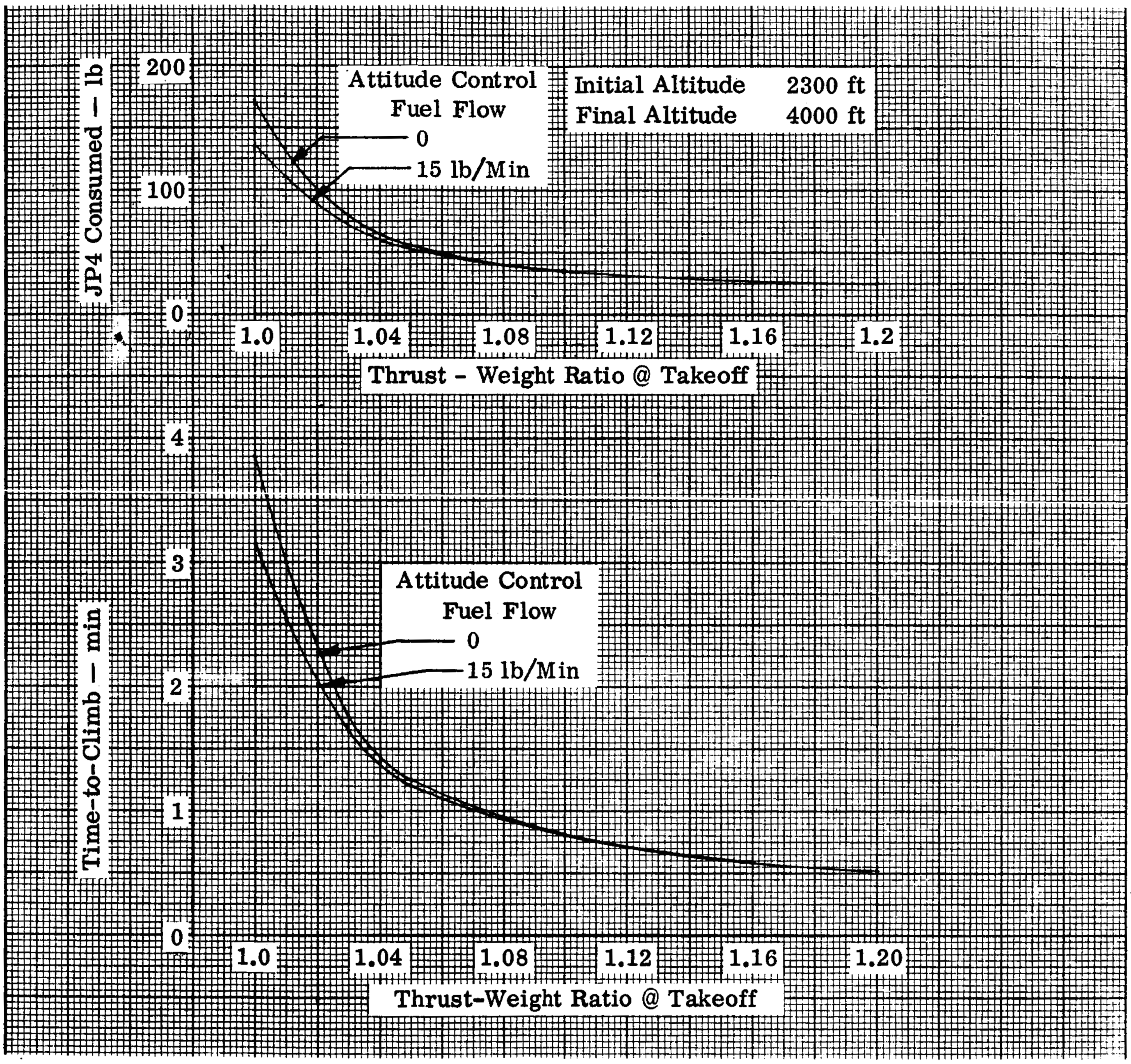


Figure 3-6. Fuel Required and Time to Climb versus Thrust to Weight Ratio at Takeoff

3.4.2. Gimbals Free. - In the gimbal free, or lunar simulation mode, the vehicle is accelerated by the component of the lift rocket thrust when the vehicle is tilted. The jet engine is tilted automatically so that the component of jet thrust is equal to the drag. Procedures for calculation of the fuel required for this portion of the flight have been discussed in Paragraph 3.2.2.

3.5. FLIGHT ENVELOPES.

The maximum altitude and translation speed boundaries are shown in Figure 3-7. Maximum altitude attainable is primarily a function of the vehicle weight and ambient temperature. Detail A of Figure 3-7 presents the rate of climb boundary versus altitude for two temperatures (standard and warm day) and two weights. These boundaries are for the limiting case of thrust equals weight plus drag.

In equilibrium translational flight, two conditions must be satisfied. The vertical component of thrust must equal the weight (or in lunar simulation the fraction of weight supported by the jet engine) and the horizontal component must be equal to the drag. Furthermore the vehicle attitude control moment must be greater than the aerodynamic moments on the vehicle (plus a margin to handle c.g. offsets, thrust misalignment, etc.). Detail B of Figure 3-7 shows the translational speed obtained versus the engine pitch angle. The graph shows that the maximum required tilt angle of the engine at a horizontal velocity of 60 feet per second is below the 15-degree maximum imposed by the engine manufacturer. Thus, the translational speed will not be limited by the allowable engine tilt.

Similarly the available attitude control moments will not restrict flight at horizontal velocities up to 60 feet per second if flight is with either the standard or test rockets or both set at full (90 pounds) thrust. If a flight is made with the attitude rockets set at a low thrust level, flight at speeds up to 60 feet per second may not be possible at all flight path angles. In such cases, speeds will be limited because the aerodynamic moments act in a direction to prevent pitching (or rolling) to attain high velocities.

Reference 4 presents additional discussion of aerodynamic moments and their effect on the operating regime of the vehicle. This reference also discusses the moment compensation system which automatically fires the attitude rockets to cancel aerodynamic moments. Because of this system, aerodynamic moments will be barely noticeable to the

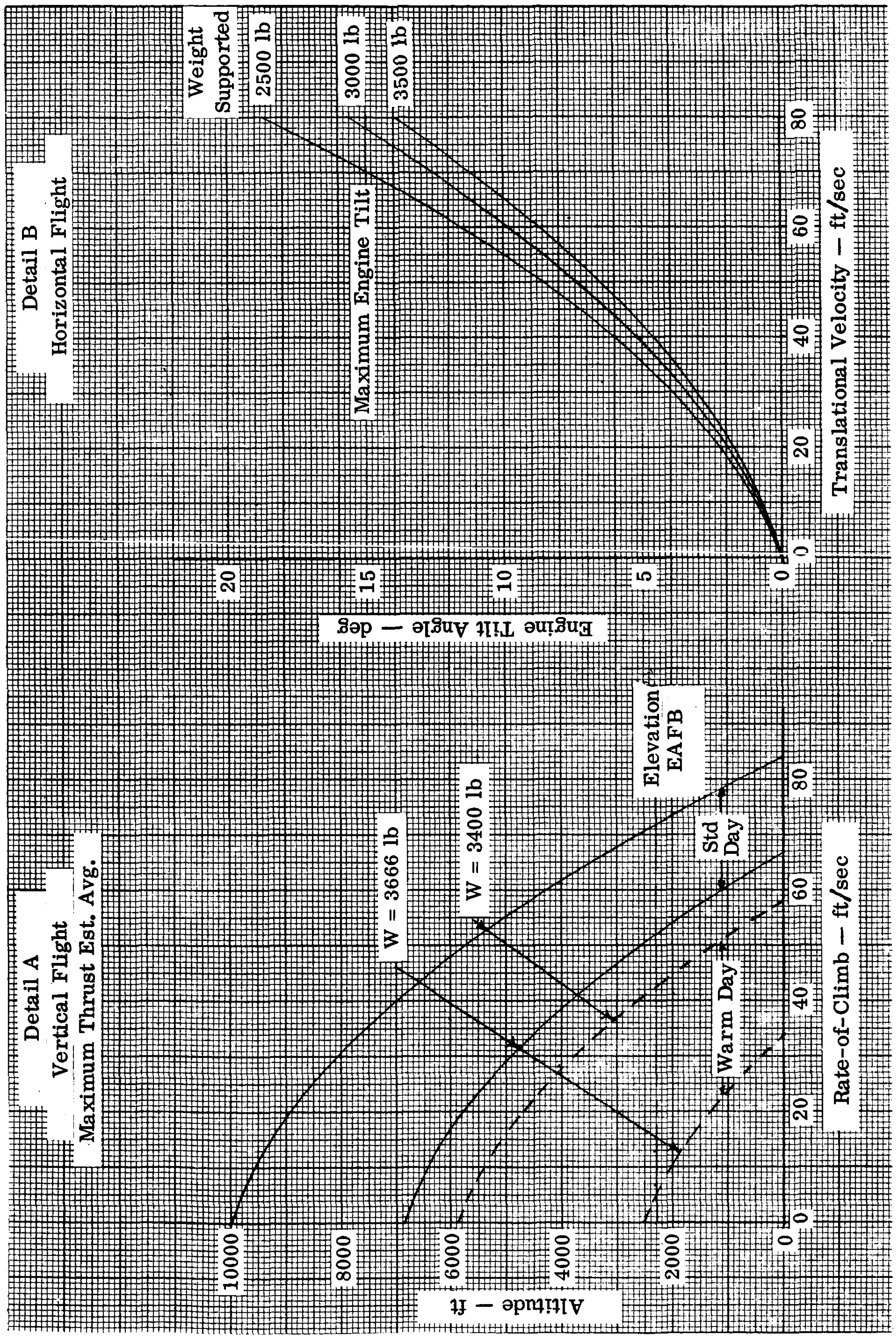


Figure 3-7. Operating Envelopes

pilot unless a large percentage of attitude control capability is required for the automatic moment compensation.

3.6. EMERGENCY DESCENT.

If a jet engine failure occurs before the lunar simulation phase of the flight has been conducted, there will be approximately 500 pounds of lift rocket fuel for the emergency rockets. Figure 3-8 illustrates the recovery profiles which can be used.

Assuming failure occurs in the free fall region, the emergency parachute must be deployed and the vehicle allowed to free fall until the descent velocity takes the vehicle past the lift rocket fuel boundary. A series of actions are now open to the pilot, the boundaries being defined by the following two cases:

(1) Continue the free fall so that the terminal descent velocity of 100 feet per second is achieved. When the altitude reaches 350 feet (for the weight being considered), apply full emergency rocket thrust and the vehicle will decelerate and touch down at 10 feet per second. Sufficient fuel is available for a 10 foot per second descent from 100 feet should the lift rockets have been applied at 450 feet instead of 350 feet.

(2) As the vehicle passes through the lift rocket fuel boundary, apply the emergency rockets to give a thrust equal to the weight and hence maintain a constant rate of descent (44 feet per second for this case) down to 150 feet. At this altitude apply full emergency rocket thrust and the vehicle will decelerate to reach the ground at 10 feet per second. This form of descent uses all the available lift rocket fuel.

Case (1) has been investigated on the analog simulator setup at Bell (Reference 4). It was found that the vehicle was controllable but that application of full thrust at the precise height was extremely difficult. If thrust was applied too early there was a distinct tendency for the vehicle to reach zero descent velocity and begin climbing again. If thrust was applied too late, the velocity of impact was too great!

This form of recovery was considered decidedly hazardous. A profile approaching case (2) would be more desirable, since the maximum descent rate is halved and hence twice as much time is available for the pilot to initiate a final braking with the emergency rockets. It will be realized however, that both the lift rocket fuel boundary (Figure 3-8) and the thrust limited boundary are functions of vehicle weight at the start of emergency descent. Thus, to perform a case (2) descent, the pilot would need to know his position relative to both of these changing boundaries. With case (1), emergency descents from

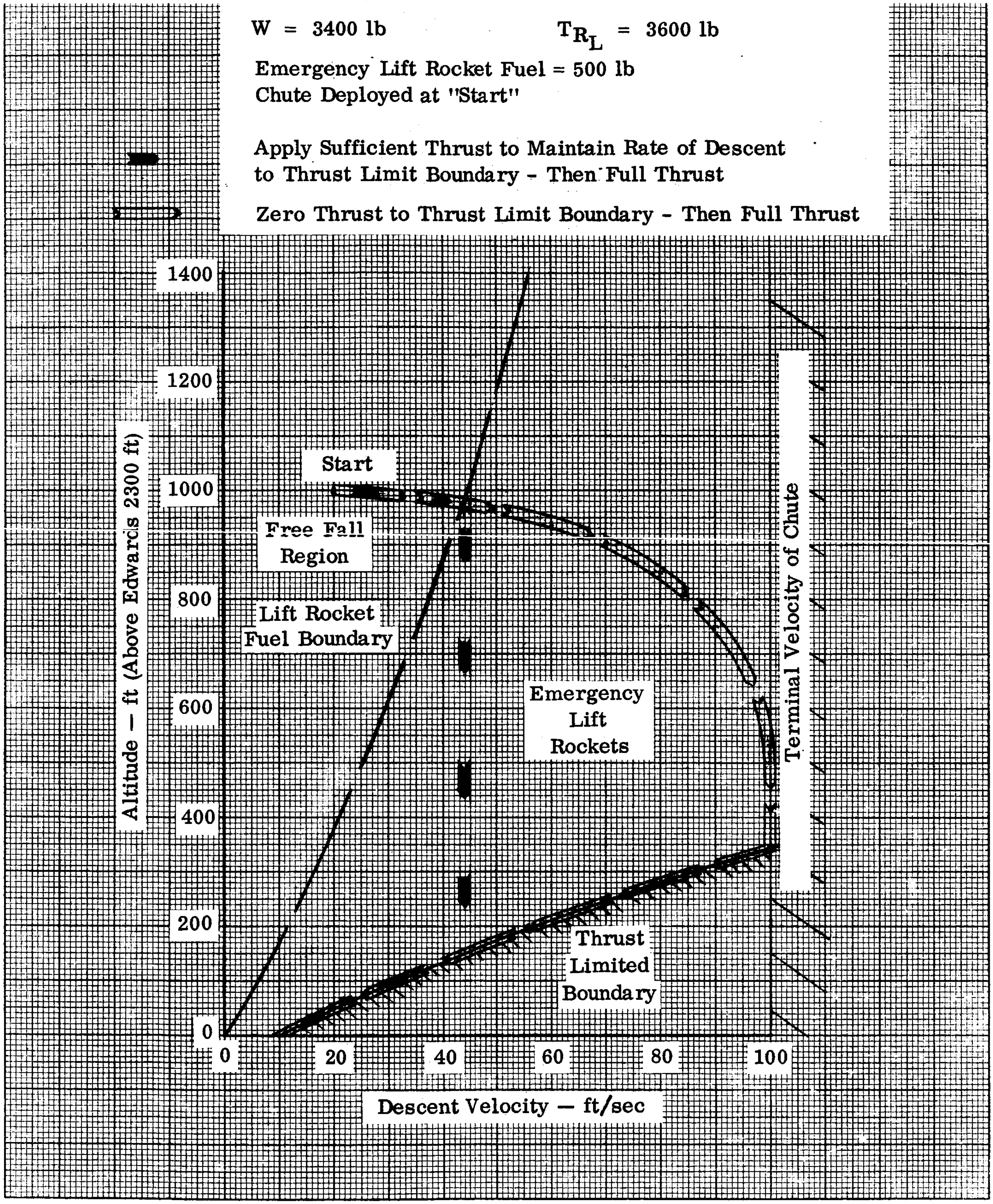


Figure 3-8. Jet Engine Failure Recovery Profile

greater than about 900 feet require knowledge of only the thrust limited boundary at 100 feet per second. Descents from less than 900 feet will have a free fall curve which intersects the thrust limit boundary at less than the drogue chute terminal velocity, so here again the pilot needs to know his position relative to the boundary.

The foregoing discussion indicates that further investigation into failure recovery is required with the intention of deriving either: (1) a standard descent procedure which the pilot can use for any weight and failure height, or (2) some form of instrumentation to indicate the appropriate boundaries. Also, it would be desirable for the pilot to have considerable practice in these maneuvers on a simulator before flying the vehicle.

SECTION IV

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